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for the period January 1, 1962 to March 1, 1962

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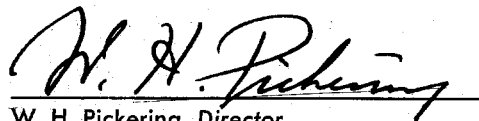
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JPL SPACE PROGRAMS SUMMARY NO. 37-14

Preface

The *Space Programs Summary* is a two-volume bimonthly publication of the Jet Propulsion Laboratory. Research and development activities conducted by JPL having specific application to space programs under the sponsorship of the National Aeronautics and Space Administration are reported in this publication. Unclassified and classified activities will be documented in Volumes I and II, respectively.

The projects reported herein were undertaken in partial fulfillment of National Aeronautics and Space Administration Contract NAS 7-100.



W. H. Pickering, Director
Jet Propulsion Laboratory



Space Programs Summary No. 37-14
Volume II (U)

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PART ONE RÉSUMÉ

I. Lunar Program

At 15:30:11 EST on January 26, 1962, *Ranger III* was launched from Cape Canaveral into a heliocentric orbit. The spacecraft executed a mid-course maneuver on command, passed by the Moon at a distance of 22,800 mi, and was tracked to a distance of 560,000 mi by the DSIF. A failure in the spacecraft computer and sequencer during the lunar flyby prevented the recovery of TV pictures.

The *Ranger* RA-4 spacecraft and launch vehicle are at AFMTC being prepared for the next launch attempt.

II. Planetary-Interplanetary Program

Both flight spacecraft for *Mariner R* have been assembled and are in the process of electrical checkout. All major aspects of the project are proceeding on schedule.

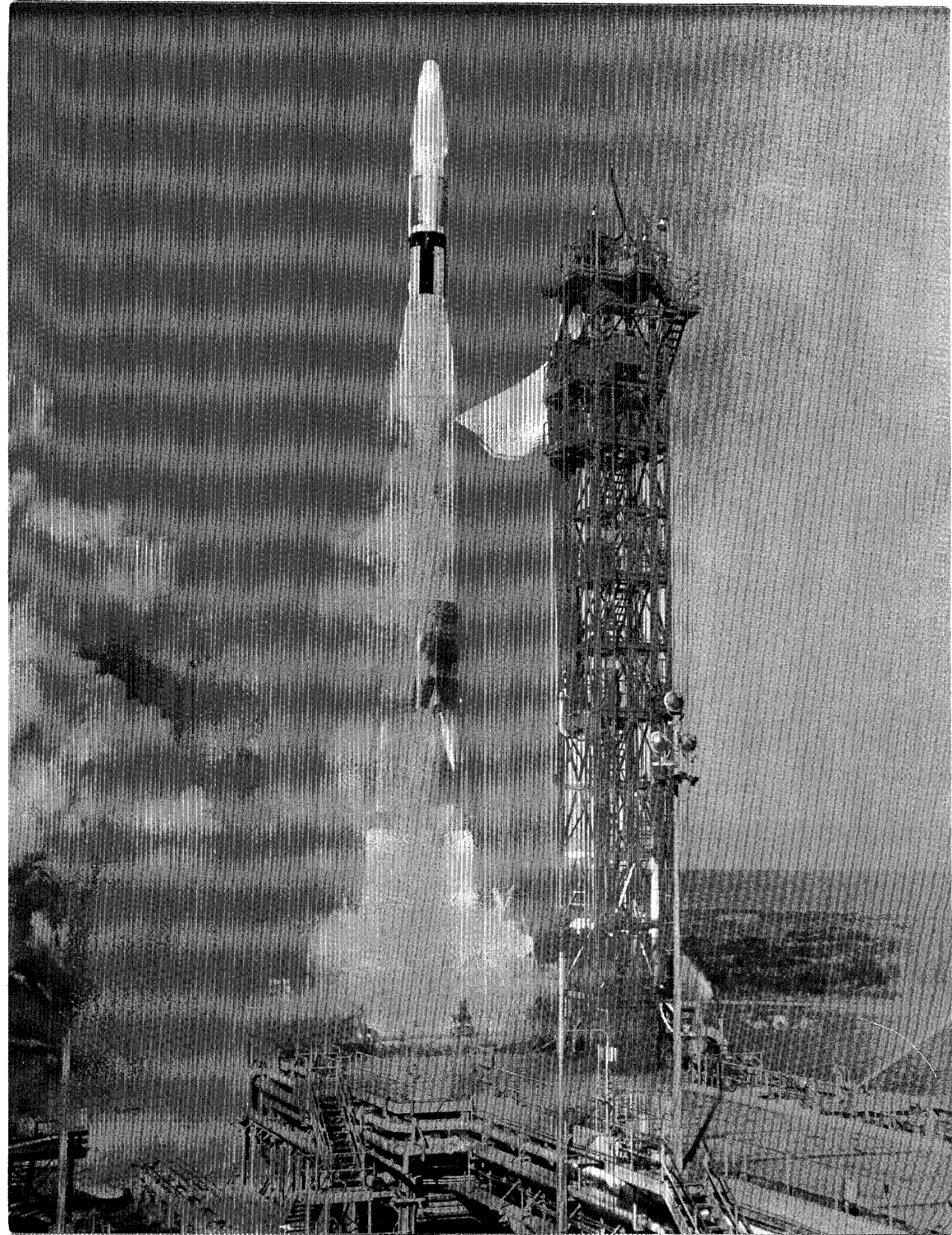
Prerelease version of *Mariner B* Project Development Plan (PDP) has been completed and forwarded to both NASA Headquarters and Marshall Space Flight Center for coordination.

III. Deep Space Instrumentation Facility

All stations participated successfully in tracking and receiving telemetry from *Ranger III* until its signals were diminished following the terminal maneuver.

Recent improvements to the DSIF include the addition of parametric pre-amplifiers, stable oscillator control of transmitter frequency, and high power L-band antenna feeds. A helical tracking antenna, as an aid to acquisition, was utilized at the South Africa station with *Ranger III*.

Research and advanced development for the DSIF continued in the fields of ranging, low noise systems, X-band communication components, receiving techniques, antenna structures, and the Advanced Antenna System (AAS). A design for the AAS has been chosen specifying a diameter of 210-ft. Supporting studies for the large antenna were made with wind tunnel tests, servo and dynamic tests, servo simulator studies, and wind measurements at the site.



PART TWO

LUNAR PROGRAM

I. Ranger Project

The object of the *Ranger III* flight was to hit the Moon during a period of a few days around third-quarter phase in January 1962 when lunar lighting would have been good for photography and a near-vertical impact trajectory could have been achieved; the usable days were January 22 through 26. Range-safety and tracking azimuth constraints established a firing window of about 1 hr on each of the 5 days.

Preflight activities continued on schedule until January 19, when during *Atlas* fueling, the fuel tank insulation bulkhead failed. At first it appeared that the mission would have to be abandoned until the February lunar opportunity. But *Atlas* personnel, working around the clock, made an ingenious and unprecedented repair without removing the vehicle from the pad. The flight was rescheduled for January 26, and the spacecraft TV camera was adjusted so that January 27 could also be used if necessary, even though lunar lighting would be poor; the MA-6 *Mercury* flight was also scheduled for that day.

Ranger III was launched on January 26, 1962, at 15:30:11 EST (20:30:11 GMT). A failure in the *Atlas* ground guidance system resulted in a late booster cutoff with no control over the sustainer cutoff time to compensate for the excess velocity accumulated. Two programmed *Agena* burning periods followed and the spacecraft was injected into an orbit with too high an energy for it to intercept the Moon. The excess injection energy was considerably greater than that which the mid-course propulsion system was designed to correct, so that the possibility of a successful lunar impact mission was ruled out early in the flight.

The over-all flight performance of the *Ranger III* spacecraft was quite successful, but there were a few

failures which have resulted in some modifications to the *Ranger* RA-4 spacecraft and supporting operations. The stations of the Deep Space Instrumentation Facility (DSIF) provided tracking and telemetry coverage continuously for 5 days except for periods when the RF signal was too weak to recover. The spacecraft signal was essentially lost at 18:54 GMT on January 31, shortly after the supply of gas used for attitude stabilization was exhausted.

A. Systems Test and Operation

1. Ranger III

From arrival of *Ranger III*, on November 10, 1961, at the JPL hangar until the launch countdown, all activities with respect to the preparation, prelaunch, and final launch countdown took place within the anticipated operations schedule (Fig 1). The *Ranger III* flight scheduled for January 22, 1962, was postponed until January 26, in order to rework a damaged isolation bulkhead in the *Atlas* fuel tank. The *Ranger* flight vehicle (*Ranger III*, *Agena* B-6003, *Atlas* 121D) was launched during the first countdown on January 26, at 15:30:11.439 EST from Complex 12 at AMR. At launch all spacecraft subsystems were in operating condition.

a. *Prelaunch operations.* *Ranger III* arrived at AMR on November 20, 1961. A complete receiving inspection did not disclose any spacecraft damage; however, several pieces of ground support equipment showed minor

shipping damage. The systems test complex cabling was started November 20; no major difficulties were encountered. The gamma ray GSE was modified to the extent that a new rack containing a punched tape and Sanborn recorder was added.

During initial systems test preparation the mid-course motor fuel tank was charged to 150 psi, and the helium tank was charged to 500 psi. The pressurization was an attempt to gain a long term evaluation of tank leak rates. Several subsystem problems developed during the test periods.

During initial power turn-on at AMR several discrete failures were noted on the data display equipment. Some of the analog outputs to the central recorder failed during testing and were repaired. The Channel V discriminator had failed prior to AMR preliminary system test (spacecraft System Test 5). The spare was not received at AMR in time to provide information during the systems test. A short circuit in the data display console prevented the data encoder from controlling the decommutator inhibit function.

Flight countdown procedure for *Ranger III* started at 09:30 EST on January 26, with an intercommunications checkout of all stations. The spacecraft was in flight configuration.

The spacecraft flight countdown was scheduled to start at 10:45 EST with time at $T - 205$. Shortly before the scheduled start time, a 25-min hold was reported to be scheduled for $T - 205$. This hold was later extended an additional 10 min. While holding at $T - 205$, the RF and telemetry trailers lost critical power. An investigation indicated that additional loading of television equipment recently added to the critical power lines was the source of the power failure. Spare generator power was used for the launch operations.

At 11:16 EST the countdown was started at $T - 205$. Several times during the countdown period, RF out-of-lock conditions were noted. Personnel and hardware motion around the spacecraft were confirmed to be the source of the problem. The scheduled hold at $T - 5$ was extended an additional 5 min for LOX topping of the *Atlas*. The spacecraft was launched at 15:30 EST with all subsystems normal. Spacecraft functions were monitored from launch until the signal was lost over the horizon at $T + 7$ min, 48 sec.

2. Ranger RA-4

a. *Preflight preparation.* Mechanical assembly and alinement were completed according to the operations

schedule. Problems associated with the mechanical assembly were:

- (1) Sterilization of the bus assembly resulted in warping the structure so that Plane A (Station 500) was out of tolerance by 0.115 in. Allowable tolerance was compensated by shimming.
- (2) A wire in the ring harness was found to be parted near a splice. The damage occurred during the sterilization process.

Electronic and electrical checkout. Initial subsystem checkout, including power turn-on, grounding and wiring integrity checks, and most subsystem calibrations, were completed according to schedule. Gyro calibrations, communications calibrations, and the vidicon subsystem checkout were delayed until November 6, 1961. During these calibrations and associated testing the following difficulties were encountered:

- (1) Transponder (Serial No. 5) failed to check out due to AGC trouble, and was returned to the cognizant engineer. The unit was replaced by Transponder 26.
- (2) The antenna drive unit (7A8) had a faulty potentiometer believed to have developed during sterilization procedures.
- (3) An O-ring failed on the attitude control pressure regulator.
- (4) The autopilot electronics unit (7A5) failed.

System testing. After all subsystems were verified, System Test 1 was performed on November 8. Minor discrepancies and procedural errors were corrected.

Further subsystem testing preceded System Test 2 performed on November 20. Two problems were noted in the pyrotechnics subsystem:

- (1) The squib simulators for the altimeter antenna deployment (bolt cutters) showed an intermittent open circuit when wiring was moved and an intermittent short to frame ground or shield under similar conditions.
- (2) Minor mechanical problems and calibration of data display channels required attention. Data encoder Channel 5 VCO was replaced.

System Test 3 was performed December 1. The following discrepancies were noted:

- (1) Command subsystem did not transmit Real Time Commands 5 and 6 properly, due to the alert pulse not counting exactly.

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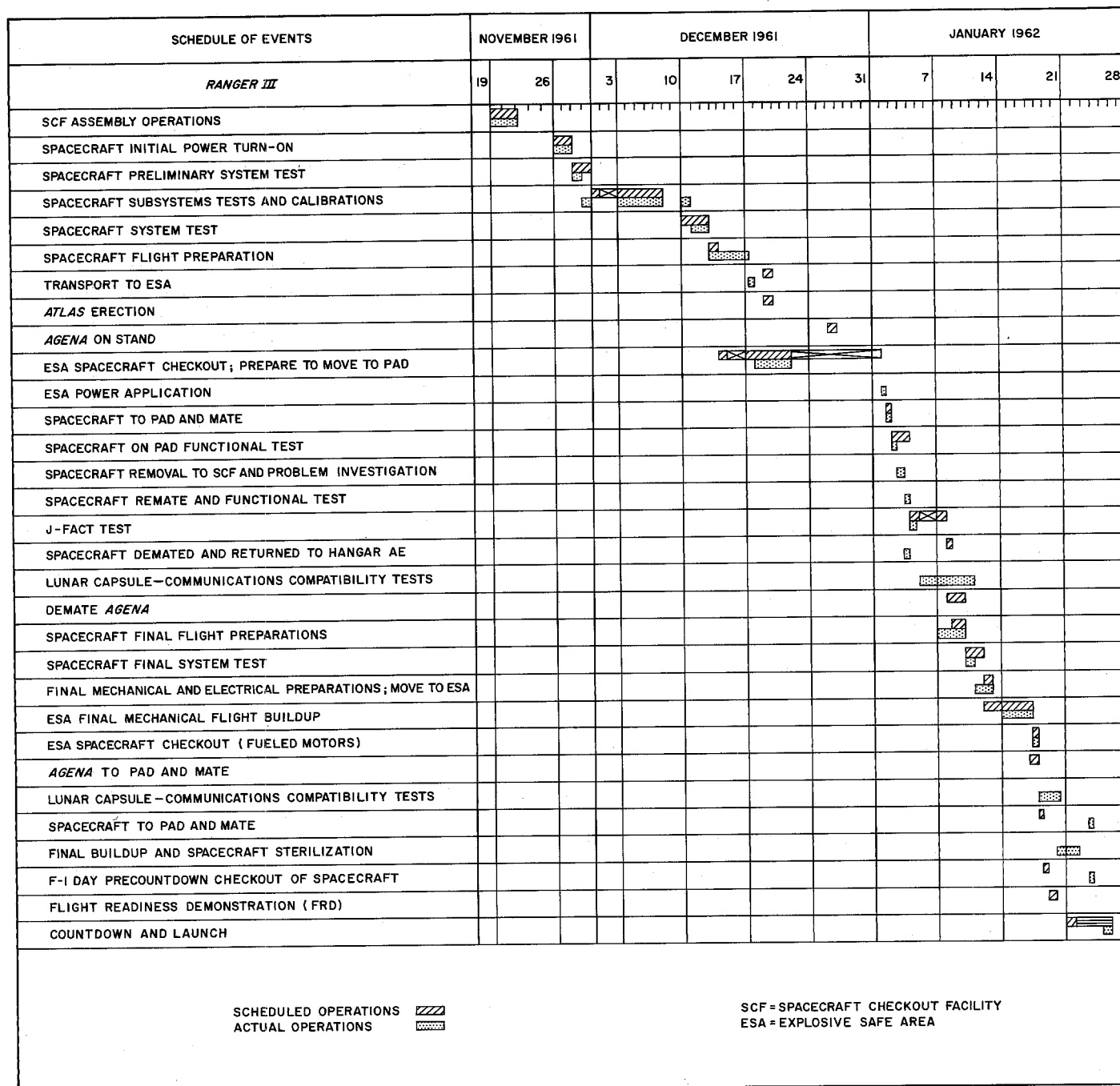


Figure 1. Scheduled prelaunch test operations at AMR

(2) Gamma ray readout sequence was interrupted due to a high voltage lead becoming disconnected.

Environmental life tests. Ranger RA-4 was readied mechanically for environmental life tests in the space simulation chamber. These tests were scheduled to begin December 7, but chamber test pressure was not adequate.

Verification tests were performed at an environmental vacuum pressure of 5×10^{-6} mm Hg, and successively stabilized at 85, 55, and at 115°C. Subsystem verification tests at these three temperatures showed that the gamma ray experiment was operating at too high a temperature, and that Channel 3 VCO of the data encoder data was noisy.

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Mission verification tests. The mission verification test, which simulates the environment in time and space for the RA-4 lunar mission, was performed subsequent to the high temperature tests. All subsystems performed within tolerances for the launch, mid-course maneuver, cruise, and encounter phases of the mission. Post-mission checks were also satisfactory. The testing was concluded by December 15.

Three precountdown (dummy run) checkouts and a verification of all spacecraft systems using live pyrotechnics were accomplished before January 1, 1962, and the RA-4 spacecraft was readied for final system tests prior to shipment to the AMR. The shipping date was scheduled for February 19. Meanwhile special GSE calibrations and subsystem testing were undertaken in preparation for the launch operations.

System Test 4 was performed January 18. The spacecraft was in flight configuration except for the following:

- (1) Command detector (3A1, Serial No. 9).
- (2) Lunar capsule.
- (3) Altimeter.
- (4) Gamma ray boom.
- (5) Gamma ray temperature sphere.
- (6) Omni-antenna.
- (7) High gain antenna.

This system test was completed without any discrepancies noted.

Between System Test 4 and the final system test at JPL, the following tests were performed:

- (1) Pyrotechnic capsule separation.
- (2) Vidicon verification data.
- (3) DSIF spectrum analysis (Mode 4).
- (4) Nitrogen pressure calibration.
- (5) Data encoder calibrations.

The final system test was performed satisfactorily on February 8.

Ranger RA-4 was shipped on February 19, to Cape Canaveral, and arrived the morning of February 27.

Spacecraft assembly followed during this and the following day, when the RA-4 spacecraft was mounted in

the system test fixture and cabled to the GSE. Complete grounding checks were performed on all GSE and associated cabling prior to mounting the spacecraft in the stand.

The RA-4 spacecraft is now undergoing initial power turn-on and preliminary system test at AMR. Integration match-mate tests with the *Agena* vehicle and the Lockheed adapter are scheduled.

b. Operational responsibility. Figure 2 shows the JPL-AMR Ranger RA-4 organization; Figure 3 identifies the JPL RA-4 test direction team areas of responsibility. This team will direct and accomplish the preparatory operations with the support of cognizant division engineers assigned to the individual subsystem groups. The test direction team is cognizant of all JPL-AMR Ranger operations associated with the spacecraft. Figure 4 shows the functional relationship of the test direction team in the Ranger RA-4 launch and space flight organization. During the combined Ranger operations, the JPL test director functions as part of the launch organization.

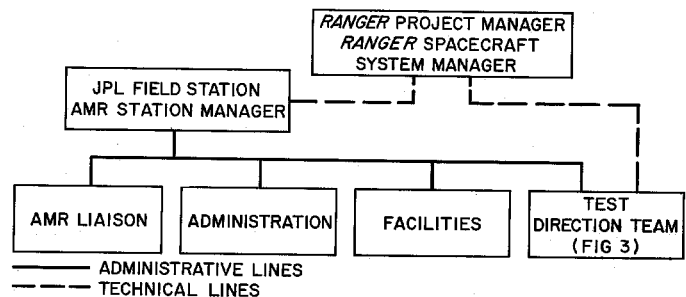


Figure 2. JPL-AMR Ranger RA-4 organization

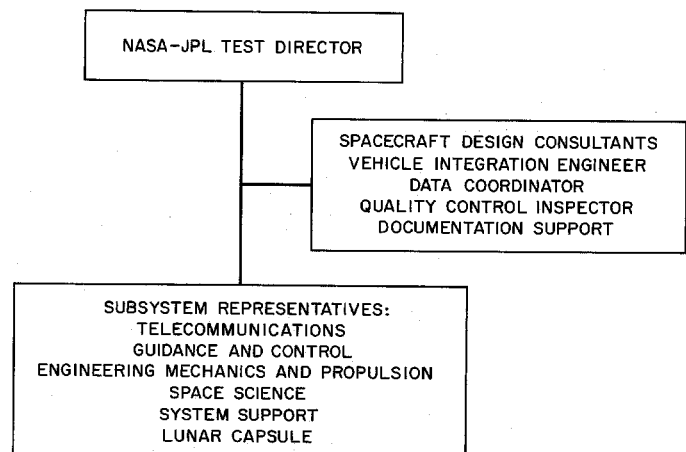


Figure 3. JPL test direction team areas of responsibility, RA-4

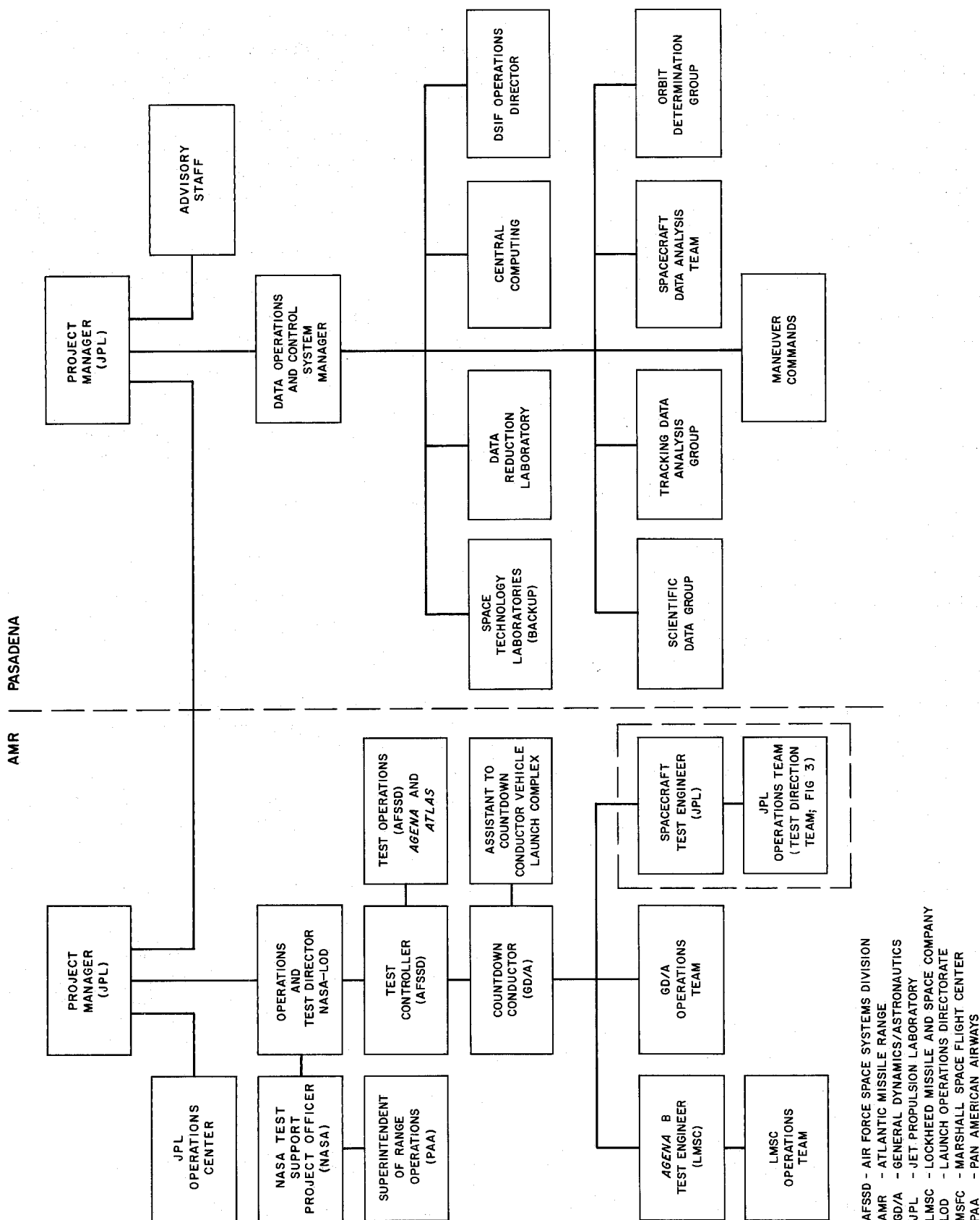


Figure 4. Launch and space flight organization

B. Telecommunications

1. Ranger Data Encoder GSE

The basic system for encoding telemetry data in a form suitable for modulation of the RF carrier for *Ranger* spacecraft is a PAM/FM or time division multiplex/FM subcarrier method. In this system many data points are periodically sampled; the sampled data then frequency modulates an oscillator. The outputs of several FM oscillators, each being modulated with sampled data, are linearly mixed and the composite signal then modulates the transponder.

The data encoder system test ground support equipment (GSE) provides the capability for controlling and monitoring the flight data encoder in the system test complex. This capability is provided by four units as shown in Figure 5.

The encoder functions required by the system test console (STC) connect to the spacecraft through the separation connectors. Other than for power monitors, there is no direct access to the data encoder proper without breaking into the ring harness lines.

Two isolation amplifiers are used in the data encoder test complex and are mounted in the system test junction box. The mixed signal line amplifier provides isolation of the STC ground from spacecraft common. It also presents a constant input impedance to the spacecraft and constant gain over the encoder output frequency range so that the encoder VCO output amplitudes can be measured accurately at the STC. The mode indicator line amplifier prevents loading of the high impedance mode indication output lines from the encoder.

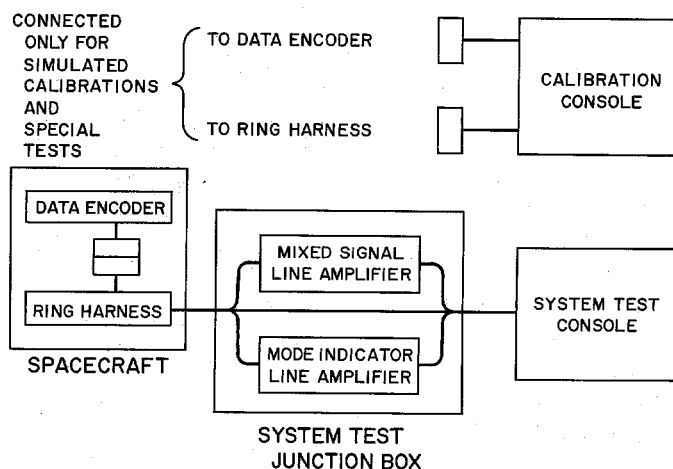


Figure 5. Data encoder components in *Ranger* system test complex

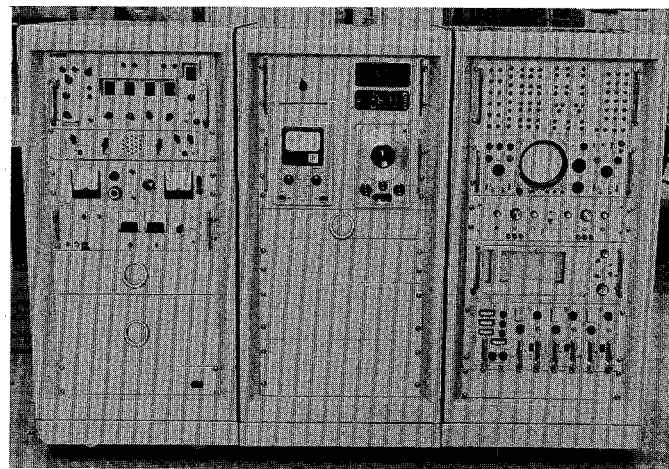


Figure 6. Data encoder system test console

The calibration console is not a standard item of system complex GSE. Its connection to the complex is made by breaking the data encoder lines to the ring harness. Since this connection is nonstandard, this unit is connected only for initial encoder checkout calibration and special testing. The circuitry of the calibration console will not be discussed in this report. The console provides for simulation of the input functions to the encoder which are not provided by the STC. It also provides the capability for terminating with the transducer impedance and measuring the transducer output.

Figure 6 shows the system test console and Figure 7 is the console block diagram. The STC receives from, and distributes signals to, several other consoles in the system complex as well as the spacecraft. The RF mixed signal is obtained from the communications console test receiver. The power line to the spacecraft from the power control unit is completed via special test cables directly to the data encoder. Since power is normally supplied from the spacecraft power converter, this connection is used only in special circumstances.

All of the standard commercial test equipment, VTVM, counters, recorder, oscilloscope, etc., in conjunction with the filter assembly, are used for mixed signal analysis. The filter assembly is a set of bandpass filters for the *Ranger* subcarriers. The patch panel on the STC allows for easy connection of the test equipment, and also provides the patch points for monitoring the data display discriminator outputs.

During normal operation in system tests the filter assembly, the VTVM, and counters are used to monitor subcarrier amplitudes and frequency. Three subcarriers may be detected by the discriminators in the STC and

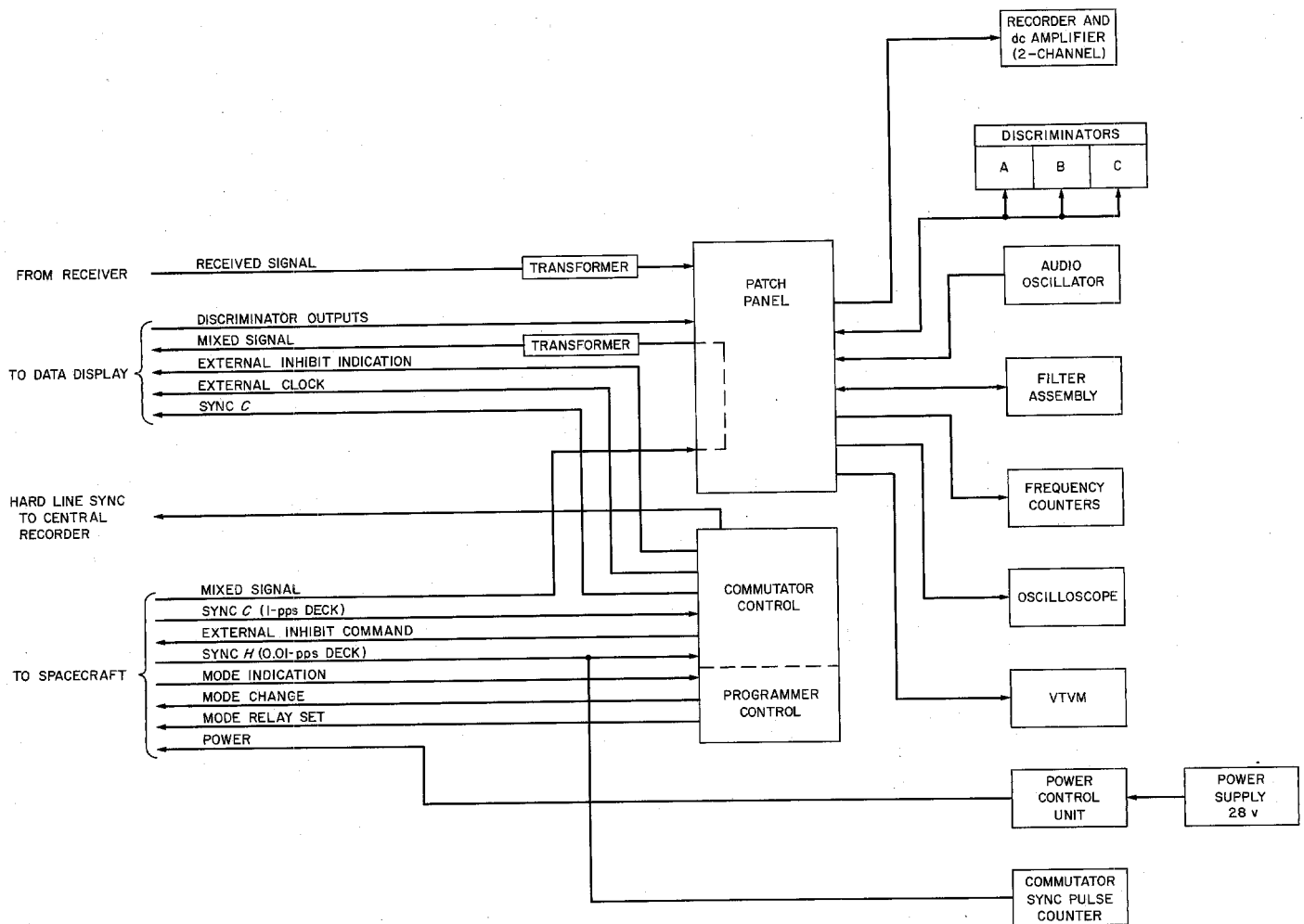


Figure 7. System test console data encoder, Ranger

the analog outputs of two of these may be displayed on the two channel recorder.

All the control functions which can be performed by the STC on the spacecraft encoder are provided by the commutator-programmer control unit (CPCU). This unit is mounted in the upper left bay of the STC and is pictured in Figure 8; the functions are shown in Figure 9.

The programmer control section of the CPCU supplies the required pulses to the encoder for changing the mode relay condition of the programmer (or data selector) module in the encoder.

The data encoder mode is indicated on the front panel of the CPCU by a small projection display unit. The indicator circuitry is straightforward, as shown in the block diagram. The input to the programmer control section is 6.8 v, which is derived from the mode indicator line amplifier discussed previously.

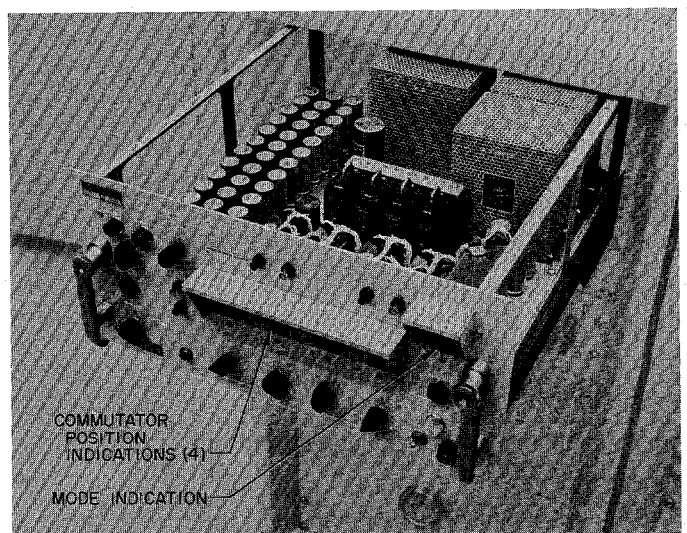


Figure 8. Data encoder commutator-programmer control unit

is done, the thrust vector will not be pointed in the desired direction in space. To correct this situation, an inertial thrust angle correction loop is added, which has the effect of rotating the entire vehicle backward through the angle of center-of-mass misalignment. The vehicle then maintains a stable attitude with its thrust vector in the proper direction.

The inertial thrust angle correction loop consists of combining the vehicle position signal with the jet vane position signal and integrating the two to drive the jet vane actuator. Since the steady-state operation requires the vehicle angle to have the opposite sign of the jet vane angle, the input to the integrator should go to zero when the vehicle position reaches its proper value.

Analysis of the autopilot includes nonlinear effects such as amplifier saturation and friction in the jet vane actuators. Calculation of these effects, however, depends on an accurate mathematical description of the actual physical situation, which may be complex and not completely known. An example of the difficulties experienced is the analysis of the effect of jet vane actuator friction on the *Ranger III* autopilot. Both mathematical analysis and analog computer simulation based on idealized coulomb friction indicated a limit cycling of the spacecraft angle. However, when the analog computer was programmed to use the actual jet vane actuators in the autopilot simulation, the expected limit cycling was not observed. The noise in the associated actuator electronics caused mechanical jitter in the actuator such that the effect of friction was reduced to a negligible level.

Analysis and evaluation of these effects upon the autopilot is aided by the single axis spacecraft simulator, which is able to simulate spacecraft pitch, yaw, or roll axes. The total autopilot system may be simulated using the spacecraft simulator plus the actual gyroscope, jet vane actuators, and autopilot electronics package. The spacecraft simulator may also be used for attitude control, acquisition, and commanded turn studies.

A block diagram of the simulator is shown in Figure 11. Torques acting on the spacecraft are represented by voltages, which are integrated electronically to obtain a voltage proportional to the spacecraft angular velocity. This operation simulates the rigid body equation

$$\theta = \int^t \frac{N}{J} dt$$

where θ is the spacecraft angle, J the moment of inertia, and N the applied torque. Nonrigid body terms arising

from structurally resonant members such as the solar panels may also be simulated here, using operational amplifiers.

The heart of the simulator is the motor-tachometer, gear train, and turntable assembly. The tachometer feedback loop around the motor forces the motor angular velocity to be proportional to the input voltage, which represents the spacecraft angular velocity. A precision gear train connects the motor output shaft with the turntable, so that the turntable angular position is dynamically the same as the simulated spacecraft. Provisions are made for mounting gyroscopes or celestial sensors on the turntable so that their actual dynamic characteristics are included in the simulation.

Figure 12 shows how the simulator would be used in an autopilot simulation. Voltages representing disturbing torques drive the spacecraft simulator, producing signals from the gyroscope. The gyroscope is connected to the actual autopilot electronics package, which drives the set of four jet vane actuators. Each actuator has a readout potentiometer producing a voltage proportional to its position. Since the torque arising from the jet vanes is also proportional to its position, this voltage represents

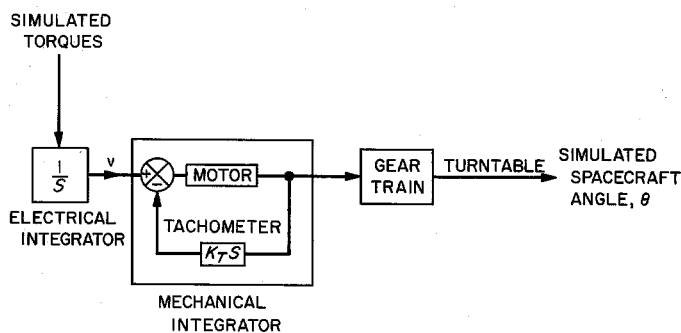


Figure 11. Spacecraft simulator

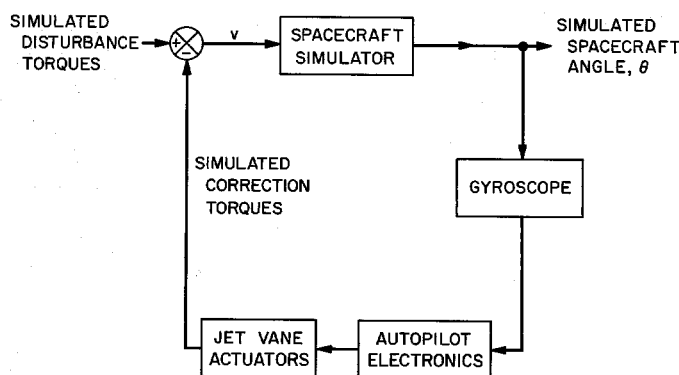


Figure 12. Autopilot simulation using spacecraft simulator

the correction torques and is scaled and fed back directly to the spacecraft simulator. In this manner, all of the nonlinear and dynamic effects of the gyroscope, autopilot electronics, and jet vane actuators are included, eliminating any estimations or approximations in their analysis.

Motor acceleration and circuitry associated with a modulator for the motor limit the frequency range over which the spacecraft simulation is valid. However, since the autopilot system works at low frequencies compared to this limitation, no problems are expected. If necessary, compensation could be added to extend the frequency range.

The question of accurate mathematical description of the spacecraft still exists, particularly with respect to structural damping and backlash in extended booms. However, shake tests on a freely suspended spacecraft enable this to be checked and modifications made in the simulation if needed.

D. Engineering Mechanics

Other than the *Ranger III* flight operations, technical efforts on the *Ranger* Project for this period have been largely concerned with the *Rangers* RA-6, -7, -8, and -9 flights.

In general, changes to the *Ranger III*-type bus structure to accommodate the TV mission have been held to the practical minimum, and a high degree of interchangeability exists with RA-6-type buses.

The *Ranger III* flight data indicate to date that the structural and mechanical component of the bus performed satisfactorily. The following extensions were noted at the correct times:

- (1) Solar panels.
- (2) Gamma ray boom.
- (3) Omni-antenna.

No verification of the radar altimeter extension was obtained although the event channel was in lock at the time.

All squib firing event pulses were recovered except the one for mid-course motor ignition. The only abnormalities associated with any of these were the transmitter failure at the time of solar panels extension, and the lack

of confirmation of altimeter extension. There was no data telemetered which would give any indication of what happened to the altimeter.

1. Spaceframe

The objective of the *Ranger* RA-6, -7, -8, and -9 series is to take TV pictures of the Moon throughout the final approach of an unimpeded impact on the Moon.

The spacecraft consists of a *Ranger III*-type spaceframe supporting a tower-like structure containing six cameras. There will be no survival capsule, retro motor, spin motor, support structure, or radio altimeter on board.

A primary design objective of RA-6, -7, -8, and -9 is to minimize changes from the *Ranger III* and RA-4 configuration. Changes have been limited to only those necessary to accommodate the TV mission and the science experiments. A few minor changes for the purpose of improving construction or reliability have been incorporated.

a. Gamma ray boom. On the TV missions there will be no gamma ray spectrometer experiment. This causes a slight modification in the solar panel latch support structure which supported the gamma ray boom on *Ranger III* and RA-4. The latch structures for both solar panels will be the same.

b. Omni-antenna. The omni-antenna installation represents a simplification. *Rangers* RA-6, -7, -8, and -9 do not require deployment of the omni-antenna. The discone antenna used on *Rangers I* and *II* will be attached to the TV payload not to be removed during the mission. Consequently, there will be no omni-antenna boom or boom actuator.

c. High gain antenna. The feed was changed from a dipole to a turnstile for *Rangers* RA-6 and -7 to provide circular polarization to the RF radiation.

Rangers RA-8 and -9 are to carry the ranging experiment which is to operate in both S- and L-frequency bands. A multi-band antenna with a conical helix feed has been designed and is being tested for RF characteristics. Since the feed and feed supports on this antenna extend further away from the antenna dish than previous designs, it has been necessary to put a jog-like bend in the pivot arm to permit the antenna to nest higher up into the spacecraft. This affords approximately 1-in. clearance between the feed supports and the *Agena* sterilization diaphragm and, consequently, interchangeability between *Agena* and spacecraft remains unimpaired.

High gain antennas of the *Rangers I* through RA-7 types have previously been test shaken while assembled on the spacecraft per type approval requirements. The shake tests to be run during the next month on the mechanical test model (MTM) will be made with the *Ranger* RA-8 and -9 multi-band antenna on board.

d. *Case III*. The half-case containing RA-4-type electronics has been eliminated along with the vidicon and telescope and midbay support member. These are replaced on *Rangers* RA-6, -7, -8, and -9 by a full Case III to contain science data conditioning modules and ranging experiment on RA-8 and -9, data conditioning only on RA-7, and transformer-rectifier and science inverter only on RA-6.

e. *Testing*. The MTM TV subsystem manufactured by Radio Corporation of America (RCA), Astro-Electronics Division, is a simulated system. Its structure, weight, over-all moments of inertia, local moments of inertia, and stiffness are the same as a flight unit. The components are weighted dummies of flight hardware. Shake tests were run at RCA on the MTM payload according to specified tests. The tests were essentially uneventful. The payload is quite stiff and appears to be compatible with the requirements.

Modal tests on MTM payload and MTM spaceframe (which is also dummied with component weights) appear to confirm that the structure is satisfactory. During these shakes a loose fastener in the camera section made some noise. One of two bolts holding a connector in place was later found to be missing. This has been corrected and all other fasteners were retightened in preparation for the type approval tests. The noise may also be attributed to a diaphragm-deflection effect of a bulkhead web which, though it is undesirable for testing, is not considered to be a serious problem.

2. Thermal Control

a. *Ranger III passive thermal control*. The passive thermal control system for *Ranger III* was a direct extension of that designed for *Rangers I* and *II*. The major thermal difference between the two types of *Rangers* is that *Ranger III*-type must survive and operate during (1) a pass through the shadow of the Earth, (2) a mid-course maneuver, and (3) a terminal maneuver. The requirement of operation during these transients made mandatory a system with minimum dependence upon solar energy. Thus, wherever possible, surfaces with a minimum solar absorptance (polished aluminum or polished gold) were selected. Also, conduction was encour-

aged within the structure and high emittance surfaces were applied throughout the interior of the thermally closed hex compartment to reduce temperature gradients during the transient periods.

Early in the program several sections of the spacecraft were built and tested under various simulated space conditions. Among these sections was a mock-up of the gamma ray detector in the extended position. Two tests were accomplished on the detector during that period to simulate spacecraft maneuvers and one test to simulate spacecraft pass through a planetary shadow.

Later as proof test model and flight hardware became available, it was possible to test the entire spacecraft hex structure (without superstructure) in the environment provided by the Laboratory 6-ft diameter space simulator.

The results of the testing program, cross checked against analysis, were used to define a pattern of high emittance surfaces to overlay the basic polished metal surface. These high emittance surfaces would reject to space the majority of the thermal energy electrically generated within the spacecraft as well as that absorbed from the Sun. White silicone paint was selected for its relatively high emittance, low solar absorptance, and ease of application. Three criteria defined the shape and location of these areas: they should (1) not be illuminated by the Sun during normal solar-oriented cruise operation, (2) be as close as possible to the heat source, (3) have a minimum view of other portions of the spacecraft.

b. *Ranger III flight results*. During the boost phase the spacecraft felt little or no effect from aerodynamic heating of the shroud. Thus, as the spacecraft entered the Earth's shadow, temperatures were essentially the same as were observed during the prelaunch phase.

Ranger III spent approximately 19 min in the shadow of the Earth. Analysis predicts a drop in solar panel temperature somewhere between 80 and 150°F, and a recovery time of some 15 to 45 min for such a shaded period. Data from this period of the flight, when available, should permit much closer temperature predictions for this portion of the RA-4 trajectory.

No engineering data was transmitted during the mid-course maneuver. In general, the spacecraft temperature seems to have increased slightly (as would be expected) judging by the data from immediately after the maneuver.

During a normal terminal maneuver, spacecraft temperature should again rise slightly. Early analysis of the data available indicates that this was not the case. Little,

if any, change was noted during the 20 min when engineering data was recorded.

The steady-state, Sun-oriented temperature control results from *Ranger III* are summarized in Table 1. With the exception of the gamma ray detector prior to its boom extension, all temperatures observed stabilized well within their tolerable limits; most stabilized within their predicted bands.

The gamma ray instrumentation temperature is the only measurement which restabilized after mid-course maneuver to a value other than its premid-course indication. During the entire portion of the flight before mid-course maneuver, this measurement indicated a gradual rise, appearing to stabilize around mid-course maneuver at 141 to 147°F.

Upon boom extension the measurement began decreasing at approximately 7°F/hr stabilizing some 7 to 10 hr later at the figure given in Table 1. Apparently, the

thermal interaction between the gamma ray instrumentation and the rest of the spacecraft before boom extension is far higher than was predicted analytically.

At present, space simulators do not have satisfactory collimation of simulated solar energy; therefore, a significant test of the spectrometer in the stowed position is presently impossible. Various optical and photographic tests are being considered, the data is undergoing further reduction, and the analysis is being refined in an attempt to design better thermal performance for the RA-4 gamma ray detector.

As the general trend of temperature measurements was toward the warm side, slightly more heat rejection area has been applied to RA-4.

c. *Rangers RA-6, -7, -8, and -9.* Since flight data on *Ranger III* has confirmed the thermal design, only the most necessary changes will be instituted. Different shadow patterns on the bus and somewhat different power levels within the cases will require surface treatment changes. These will be kept to a minimum in accordance with essential change requirements.

Table 1. Steady-state, Sun-oriented temperature control results for *Ranger III*

Location of temperature measurement	Allowable <i>Ranger</i> temperature, °F		Observed stable temperature, °F		Predicted <i>Ranger</i> temperature, °F	
	Min	Max	Prior to terminal maneuver		Min	Max
+ x solar panel front	-103	257	144	147	65	122
+ x solar panel back	-103	257	124	127	65	122
- x solar panel front	-103	257	140	141	65	122
- x solar panel back	-103	257	119	122	65	122
Gyro package	64	132	123	126	70	132
Nitrogen tank	41	158	100	104	40	85
Leg F of structure	40	158	91	95	40	85
Booster-regulator chassis	32	158	120	121	85	139
Attitude control power converter chassis	32	158	105	108	68	139
+ y yaw jet vane actuator	41	158	76	77	40	85
Earth sensor chassis	14	100	73	74	22	95
+ y pitch Sun sensor	55	115	74	78	40	85
- y pitch Sun sensor	55	115	90	92	40	85
Transponder chassis	32	158	121	123	66	113
Battery	50	131	90	92	70	105
Helium tank	41	158	90	94	40	85
Power switching and logic chassis	32	158	118	122	80	129
Gamma ray experiment	50	122	94	98	40	85

The TV subsystem thermal control will be the responsibility of RCA. The basic design calls for an isothermal shield maintained at 10°C by means of solar energy impinging on radiation fins around the thermal shield. The TV cameras are heated by the components attached to the structure within the thermal shield and have the required aluminum foil interfaces to provide good conduction heat-sink capabilities. The entire design is now being tested, using a temperature control model in a radiation chamber at RCA.

The interface between the TV subsystem and the bus will be effectively insulated by means of micarta or Epoglass shimming. These materials are now undergoing thermal and mechanical tests for best selection.

Thermal design for the ion chamber for RA-6, -7, -8, and -9 has been completed. Due to the strict thickness and surface treatment restrictions imposed by scientific instrumentation, ability to control temperature of the chamber sphere is limited. Thermal design on the amplifier electronics unit has been completed and temperature control will be held within required limits. Thermal design on the Geiger counter experiments and the dosimeter experiments to be flown in later spacecraft is well

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underway. All other thermal control provisions will follow the *Ranger III*-type design as closely as possible.

3. Pyrotechnic Subsystem

a. *Ranger III*. As reported in *SPS 37-13*, handling and checkout procedures were written and were used during the *Ranger III* assembly for the J-FACT. The procedure was revised and used during the assembly for flight, then again revised. It is felt that the present procedure, incorporating knowledge gained from the *Ranger III* operation, will serve adequately for the RA-4 assembly operation.

Telemetered data from the *Ranger III* flight gave direct indication that the pyrotechnic subsystem functioned as required in most respects. Although no indication was received that squib firing current was delivered for mid-course motor ignition, it is known from other indications that the motor did ignite.

The lack of indication of altimeter antenna deployment has not been explained. Indication was received that firing current was delivered through the squib firing assembly at the time in question. However, current is switched to a squib on the vidicon lens cover latch as well as to the altimeter antenna deployment squibs at this event; thus it cannot be confirmed that the deployment squibs were fired.

The telemetered data gave indication of another problem apparently associated with the pyrotechnic subsystem. This was the drop in signal level of the spacecraft communications system, coincident with the squib firing for deployment of the solar panels. Possible causes of the partial failure which might be attributed to the squib-actuated pin-pullers (such as mechanical shock or electrical disturbances) have been investigated.

Special tests were conducted on the *Ranger PTM* in which numerous pin-puller firings were made while monitoring the communications system. The test procedure was oriented toward investigating the effect of the mechanical shock. In summary, the results of the test indicate that no degradation in transponder or amplifier cavity performance, similar to the *Ranger III* flight data, was reproduced by firing pin-pullers either in the flight configuration or in doubles (a total of eight pin pullers). Continuing effort is being applied to analyze the nature of the shock and to determine the feasibility of dynamically isolating the pin-pullers to attenuate the high frequency shock components.

Investigation of the effect of electrical disturbances has been made before and after the failure. Other than transient efforts, no communications system degradation has been noted.

b. *Ranger RA-5*. The squib firing assembly for RA-5 will have functional capability for a lunar capsule mission with the additional requirement to fire the dual bridge-wire squibs on the mid-course motor valves. Furthermore, this unit design can be used directly for the TV mission functional requirements.

c. *Rangers RA-6, -7, -8, and -9*. The only pyrotechnic functions required for the TV missions are (1) solar panel deployment, (2) mid-course motor ignition, and (3) mid-course motor shutoff. Because these requirements are considerably less than those of a lunar capsule mission, a firing unit of simplified design will be used on RA-6, -7, -8, and -9.

4. Equipment Packaging: *Rangers RA-6, -7, -8, and -9*

Problems in procurement of electronic hardware have necessitated certain changes in radiation experiments to be incorporated. Present plans are to include the Neher ionization chamber on all flights. The particle flux detector is to be phased into RA-7, -8, and -9. Finally, on RA-8 and -9 the tissue equivalent ionization chamber will be included. The electronics associated with each of these assemblies includes a transformer-rectifier (T-R) signal conditioner and inverter to be used with the Neher chamber and housed in Case III for RA-5 and -6. A data conditioning system to be used with all experiments is to be located in Case III, and a T-R and inverter is to be located in Case VI for RA-7, -8, and -9.

The particle flux detector (*SPS 37-13*) is to be identical with the instrument to be flown on *Mariner R* with the following exceptions. Repackaging of this instrument was required in view of different mission requirements (lunar vs planetary mission). Repackaging was also required in order to accomplish a coincidence measurement of radiation in the vicinity of the Moon; therefore, tube orientation was laid out in a manner to accomplish this. Aside from additional electronics required for the coincidence circuitry, the electronics will remain the same.

The Neher ionization chamber is the same as that of *Mariner R*. Its mounting is to be beneath Foot D of the spacecraft, just above the high gain antenna.

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Currently, the tissue equivalent ionization chamber is being re-examined as to its utility as a radiation experiment. This involves the possibility of repackaging the experiment into individual chambers, such as the Neher ionization chamber, instead of manifolding all chambers into one chassis as in the present version. This would require that the electronics be incorporated into each chamber and individually mounted to the spacecraft. The chambers, by necessity, must still mount in the same area presently set aside, i.e., under the hex of Bay III and between the hex and the high gain antenna.

5. Terminal Sterilization

Sterilization operations for *Ranger III* started on January 1, 1962 and continued until $T - 180$ on launch day, January 26, 1962, and were divided into two phases.

- (1) Terminal sterilization.
- (2) Maintenance of sterile condition until launch.

a. Terminal sterilization. Final terminal sterilization operations were performed from 17:20 EST on January 23 to 09:20 EST on January 24, 1962. The required conditions were an exposure to a minimum concentration of 21 vol % ethylene oxide for 11 hr. The actual minimum conditions are shown in Table 2.

Table 2. Minimum conditions for final terminal sterilization operations

Time		Concentration, %	Relative humidity, %
EST	Min		
January 23, 1962			
2000	0	22.2	49
2030	30	24.2	53
2130	90	24.9	56
2230	150	21.2	45
2330	210	25.8	56
January 24, 1962			
0030	270	25.6	55
0130	330	22.3	50
0230	390	22.4	48
0330	450	22.2	47
0430	510	22.2	47
0530	570	22.4	46
0630	630	23.4	46
0700	660	24.2	49

Following the 11-hr exposure period, the ethylene oxide was purged with dry nitrogen, and passed through a sterile biological filter until 0% ETO and 10% relative humidity values were obtained.

b. Maintenance of sterile conditions until launch. Upon completion of sterilization operations, dry nitrogen gas was slowly introduced through a sterilized biological filter and into the spacecraft assembly. This slow flow was continued during transport to the gantry and was not discontinued until $T - 180$ of the prelaunch countdown. Primary reasons for the slow flow of sterile gas were to minimize relative humidity in the spacecraft assembly, and to eliminate any residual ethylene oxide not previously purged from poorly vented areas in the spacecraft.

E. Propulsion

Accomplishment of the lunar impact mission of *Ranger III*, RA-4, and -5, and the high-resolution television investigation of the lunar surface and terrain on *Rangers* RA-6, -7, -8, and -9 requires the spacecraft to undergo a single mid-course propulsion maneuver to remove or reduce Agena injection dispersion errors. This maneuver is accomplished through the use of a small, monopropellant-hydrazine-fueled propulsion system of 50-lb vacuum thrust, which is capable of delivering a variable total impulse in conjunction with an integrating accelerometer system.

All valving functions on the propulsion system are accomplished through the use of explosively actuated valves. On *Rangers* RA-5, -6, -7, -8, and -9, recently developed dual-bridgewire hermetically sealed squibs will be used in the explosive valves in lieu of single-bridgewire squibs to attain a higher degree of reliability. A more detailed description of the propulsion system, the operational design philosophy, and a system schematic are contained in SPS 37-3. Developmental progress is reported in SPS 37-3 through 37-13.

Project activities during this reporting period have centered upon completion of flight acceptance testing of the components required for buildup of the RA-5 and

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RA-5 spare mid-course propulsion systems. Assembly of the propulsion systems has begun in preparation for incorporation in the flight spacecraft in accordance with the spacecraft assembly schedule. The RA-4 and RA-4 spare propulsion systems have been heat sterilized and leak checked prior to their shipment to AMR for the RA-4 launch.

Initial testing with the dual-bridgewire, hermetically sealed squibs has disclosed a design deficiency in the trigger assembly, the part in which the squib is attached. Upon firing the squibs in the valve, yielding and sometimes rupturing of the trigger assembly occurred. However, rupturing of the trigger assembly occurred only when the proper techniques for assembling the squibs in the trigger assemblies were not followed. The solid propellant charge for the dual-bridgewire squib is not the same as that used in the single-bridgewire squib; thus it is felt that the failure can be attributed to the different pressure-time characteristics of the dual-bridgewire squib. The failure was in tension; it is felt that increasing the wall thickness of the trigger assembly will solve the problem.

Although the *Ranger III* flight did not fulfill all its mission objectives, it is apparent from telemetry data that the mid-course propulsion system performed satisfactorily during the command mid-course maneuver sequence.

Prelaunch preparation of the mid-course propulsion system at AMR consisted of two phases. During the first occasion when the unit was dismounted from the system and made available, the unit was subjected to a complete visual inspection, and the tankage and plumbing checked for leaks. The unit was then placed in a sterilization oven and the entire assembly subjected to 257°F for a 24-hr period. Upon removal and cool down, the unit was again leak checked. At this time a minute leak was found to exist in the propellant tank bladder. This difficulty necessitated dismantling the tank and substituting a spare bladder. After verification of the repair, the tank and bladder assembly were resterilized. No additional problems were encountered and the unit was returned to the spacecraft for a final systems test.

Several days later the unit was returned for final preparation, fueling, and pressurization. Loading and pressurization of the ignition system proceeded without incident. However, when the fuel tank was loaded with anhydrous hydrazine, fuel leakage across the propellant tank bladder was noted. Inspection indicated no leaks in the bladder itself but pointed to the probability of a leak occurring at the point where the bladder is attached

to the tank outlet fitting. The difficulty stemmed from a change made late in the development program which increased the amount of fuel loaded in the tank from the initial design value of 13.5 to 14.6 lb. This change (which was desirable but not mandatory), was made as added insurance to take advantage of an increased weight allowance available in the spacecraft in order to furnish an excess in maneuver capability beyond the design criteria. Details of this change may be found in SPS 37-11. Preliminary tests indicated that the change could be accommodated without difficulty. However, upon close inspection in the field it was concluded that the bladder wall thickness in the production hardware was so close to the upper tolerance limit (a condition not present in the prototype hardware used in the preliminary tests) that sufficient expansion was not possible without deforming and reducing the bladder neck wherein the bladder was attached to the tank outlet fitting.

It was found necessary to return to utilizing the original quantity of fuel specified. However, this still produced a velocity increment greater than the original design point, inasmuch as the over-all spacecraft weight was less than that of the initial design. In the final operation, the system as loaded had a velocity increment capability 12.5% greater than that of the original design, rather than an anticipated 20% greater capability.

The remainder of the loading and pressurizing sequence proceeded without difficulty. One additional change was made wherein nitrogen gas was substituted for helium gas in the reservoir to take advantage of the excess weight factor available by increasing gas leakage reliability.

After verifying unit pressure integrity, the system was installed in the spacecraft. It is interesting to note at this point that the fueled and pressurized unit remained in the craft for 12 days prior to launch. During the countdown, readiness of the system was verified by monitoring the pressures and temperatures of the gas reservoir and the fuel tank.

Inflight telemetry data indicated no degradation in pressure levels or change in temperatures occurring in the system for approximately the first 4 hr. After this time it was noted that the entire spacecraft was gradually rising in temperature. The telemetry data recorded prior to and in flight are shown in Figure 13. This increased temperature condition is mirrored in the gas reservoir pressure which rose somewhat. Fuel tank pressure did

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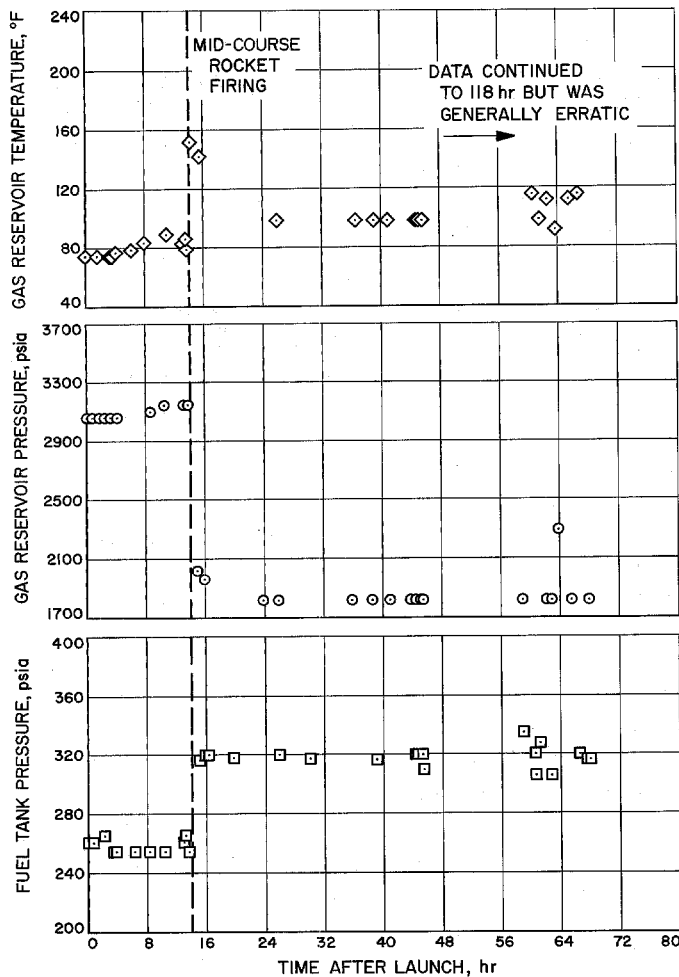


Figure 13. Telemetered mid-course propulsion system data, Ranger III

not show a comparable increase, but this is felt to be due to its more central and isolated position in the spacecraft.

The mid-course maneuver sequence was undertaken approximately 13½ hr after launch, with the actual rocket firing occurring 27 min after the start of the maneuver positioning. From post-firing data it appears that the rocket fired and shut down in a normal manner. Unfortunately, no telemetry data during the mid-course sequence was obtained because of radio transmission problems; thus, the details of the actual rocket operation must be deduced from data taken before and after the maneuver.

After firing, the pressures and temperatures appear to be as expected, with the gas reservoir temperature showing the effects of thermal radiation from the rocket engine. Although data obtained for the next several days were erratic at times, it appears as though the tank pressures remained constant indicating the continued leak tightness of the system.

The exact velocity increment produced by the propulsion system could not be determined from the telemetry records. However, it was noted that the duration of the tracking station recordings of the amount of the doppler shift during the firing of the rocket agreed quite closely, within a few tenths of a second, with the anticipated rocket burning time based upon nominal performance values obtained in the development program. Thus, it would appear that the actual specific impulse of the unit was close to the design value.

II. Ranger Lunar Capsule

A. Summary

Under subcontract for development of a complete lunar capsule subsystem, Aeronutronic Division of Ford Motor Company (ADF) is supplying a lunar seismographic capsule and ancillary equipment for *Rangers* RA-4 and -5. The *Ranger III* capsule subsystem was delivered on schedule with no significant deviations from specifications.

Included in the subsystem are the following components: an altimeter for sensing the proper instant of separation; a spin rocket motor to provide stabilization during the retro-rocket braking; a retro-rocket; a crushable impact limiter to absorb residual energies; and the survival package. The survival package is a sphere which is fluid floated inside a hollow, spherical impact limiter to distribute the structural loads of impact and to allow erection by Moon gravity to the local vertical after the assembly comes to rest.

As the *Ranger* spacecraft approaches the Moon, the capsule is to be separated from the spacecraft and braked to allow a survivable landing which emplaces a sensitive seismometer on the lunar surface. Data from the seismometer is to be relayed back to Earth receiving stations for an extensive period of time after each landing.

Conventional chemical batteries supply the total input power demand. Near-constant temperature for the sur-

vival sphere equipment is provided by the use of a cryogenic insulation and a small quantity of water which cools by evaporation to the vacuum surroundings during the lunar days. The water also offers a safety feature for night operation by virtue of its latent heat of freezing.

Considering the major elements external to the landing sphere, the altimeter was completed and delivered on schedule by Wiley Electronics. The retro-rocket program proceeded well. Previously encountered failures were successfully diagnosed, and the corrections confirmed by test firings. A nozzle deterioration, noted in the last 2 sec of vacuum burning, was shown not to cause a meaningful degradation of performance. Although the nozzle was acceptable for flight, it was decided to undertake a nozzle improvement effort. This program may provide an improved nozzle for subsequent flights. The support structures, squibs, clamps and associated equipment, and the power and sequencing assembly are completed and available for all flights.

In this reporting period three flight spheres were assembled. Two of these were sent to AMR, the third was lost due to an internal malfunction. Some product improvement efforts were started and were underway to provide a still greater confidence level for the RA-4 flight articles. These include an improved spinup mechanization and an improved retro-rocket nozzle. Tests are planned of the sphere which was not flown.

B. Test Activities

The test activities covered in this reporting period include *Ranger III* flight preparations, testing, and launching at AMR and a spin motor test effort.

1. Ranger III Joint Flight Acceptance Test at AMR

ADF support of flight preparations started at AMR on November 29, 1961, with efforts directed toward assembly and checkout of the electronic simulator to be used in the preliminary system test; the simulator confirms the integrity of all bus cables transferring commands to the lunar capsule. This test was completed successfully December 4.

Positioning and assembly of the Gisholt dynamic balance machine was completed on December 8. Checkout was completed and the lunar capsule balancing fixture was installed and balanced. The operating console for the balancing machine was installed in a remote position. A closed-loop television monitor was used to observe the balance operations.

The flight altimeter was assembled and installed on the bus, and performance and calibration tests completed on December 11. Fit checks of the flight retro-motor support structure and the altimeter assembly were completed December 14. The electronic simulator retro-motor and landing capsule were installed on the flight retro-motor support structure and bus for the first flight-condition fit check of the radiation shield.

Using a lunar capsule electronic simulator for the mechanical buildup of the J-FACT configuration (Fig 1) allowed an opportunity to perform all field assembly and checkout procedures, and to evaluate the adequacy of handling fixtures. The complete assembly and checkout were completed on December 22.

2. Ranger III Flight Preparation

On January 3, 1962, the spacecraft was mated to the *Agena* secondary booster on the launching pad. The flight retro-motor assembly was completed and readied for dynamic balance on January 6. At this time, the spacecraft was demated from the *Agena* and returned to the hangar to start disassembly and final flight buildup. The flight altimeter received its final check, fuzing signal setting, and flight battery installation. It was then assembled onto the altimeter support structure.

Dynamic balance of the retro-motor assembly was completed on January 9. The landing capsule (Serial No. 010) and vibration dampers were then assembled on the retro-

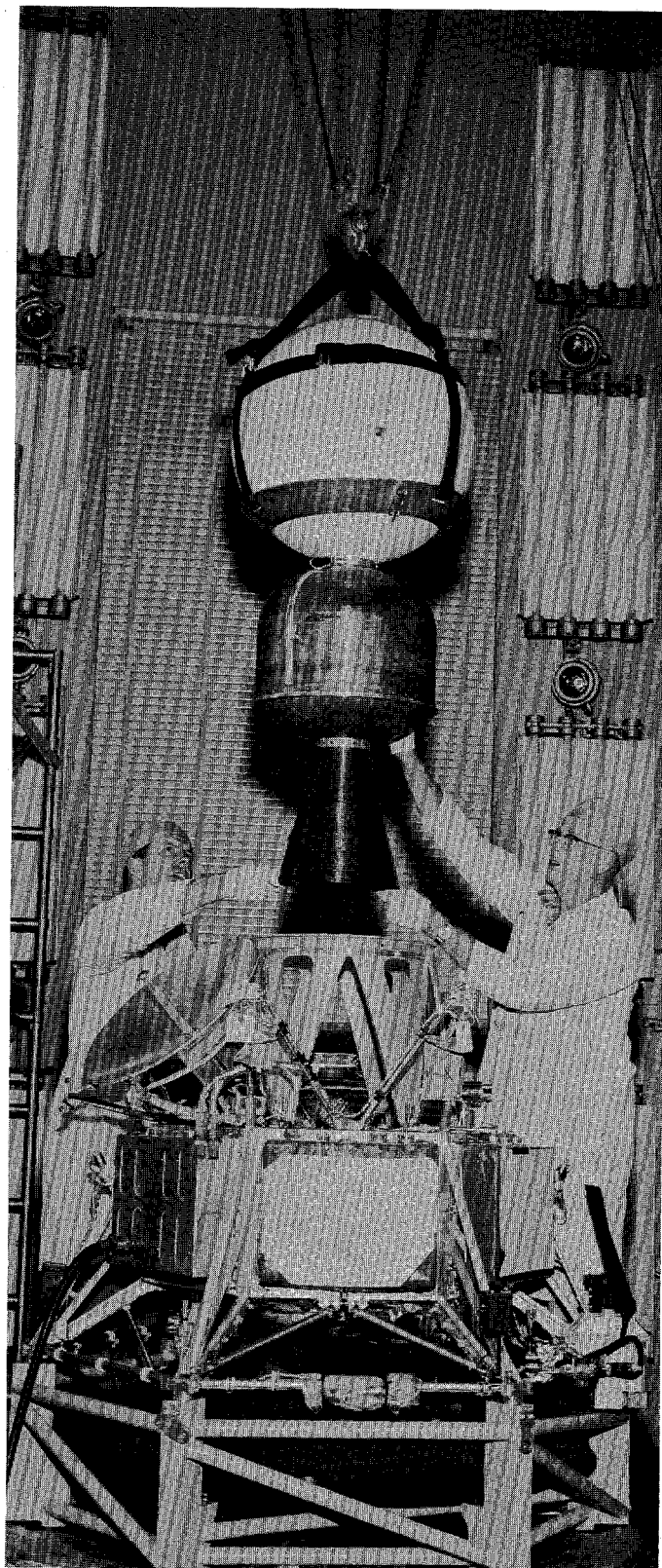


Figure 1. J-FACT buildup, sphere and retro-motor assembly

motor and dynamically balanced. The sphere was then removed from the assembly and inverted to conserve the capsule batteries. The symmetrical venting device was assembled into the flight retro support structure and again fit-checked on the bus.

On January 13 the complete lunar capsule was balanced and ready for installation on the bus. A preliminary weight was taken using the JPL load cell weighing equipment and personnel. The capsule was prepared for mating, and installation on the bus and final preparations procedure were completed on January 16.

The spare landing capsule (Serial No. 012) arrived at AMR on January 16. Spectrum frequency tests were run on the capsule to obtain comparative data on spurious RF spikes observed in the first capsule transmitter output at about 890 mc. These tests confirmed that the spare capsule power output at the transmission frequency was down 10 dbm from that of the first capsule. The spurious RF spikes at 890 mc were evident but of a lower grade.

The spare capsule was exchanged with the first capsule on the spacecraft, and RF interference tests rerun. The spare capsule was found to be free of interference and was used on *Ranger III*.

Monitoring of the *Ranger III* capsule from the ground station was accomplished as planned. The receiver stayed locked on signal for the first 7 min of the flight.

3. Spin Motor Qualification

Qualification testing of the *Ranger III* spin motor was completed in December. Flight motors were shipped to AMR according to schedule. Conditions of the qualification test firing are summarized in Table 1. Of these test firings, the spurious torque data on six tests are meaningful: Tests 114, 118, 127, 141, 142, and 148. A computer program was prepared to simulate the dynamics conditions under which the spin motor firing occurs, and the outputs of primary and spurious torques from these firings were used in the solution. The principal results of this analysis are given in Table 2.

On all except Test 148, the spurious torque exhibited a large transient during the first 400 ms of the run. Peak values of spurious torque during this transient ranged up to 3 ft-lb. After the transient subsided, the spurious torque was relatively steady for the remainder of the run, with values ranging from 0 to 1.5 ft-lb.

On the supposition that the transients were caused by shock waves or separation in the exit cone at sea level pressure, the nozzles on the motor for Test 148 were

Table 1. Spin motor qualification firing summary^a

Test	Date	Condition	Fixture
104 ^b	11-17-61	Sea level	Torque ^d
114	11-21-61	Sea level	Torque ^d
118	11-27-61	Sea level	Torque ^d
127	12-4-61	Sea level	Torque ^d
130	12-11-61	Sea level	Inertial ^d spinup
141	12-18-61	Sea level	Torque ^d
142	12-18-61	Sea level	Torque ^d
148 ^c	12-27-61	Sea level	Torque ^d
149	12-28-61	Vacuum	Special ^e

^aResults of all tests were successful.

^bMotor for Test 104 manufactured with equal-diameter exit nozzles.

^cExit nozzles truncated to area ratio of 2.25 to simulate vacuum firing.

^dTorque data obtained.

^eNo torque data obtained.

Table 2. Torque data analysis

Test	Final roll rate, rad/sec	Final tipoff angle, rad
114	27.1	0.023
118	28.3	0.019
127	26.4	0.023
141	29.1	0.030
142	28.3	0.041
148	26.2	0.003

truncated to an area ratio of 2.25. As a result, the nozzles should run full at the system starting pressure rather than at the pressure which occurs at 400 ms. The starting transient was greatly reduced on this firing, although its shape was similar to that previously experienced.

Subsequent firing tests were conducted with the same torque fixture in a vacuum chamber. The vacuum chamber is located at the Douglas Aircraft, Long Beach facility. This chamber has a volume of about 4000 ft³ and can furnish a pressure altitude up to about 140,000 ft. Prior to being installed, the fixture was mounted on the separation test fixture, which is in the vacuum chamber. This earlier mounting configuration apparently caused some dynamics response during the starting transient that coupled into the spurious torque readings; this effect has been eliminated by mounting the fixture on a stiff I-beam base, and by ballasting.

Eight motor firings were made in the vacuum chamber prior to the *Ranger III* firing. The general firing conditions are shown in Table 3.

On Tests 164 and 169, the spurious torque starting transient was again much in evidence, making it apparent that the cause was not ambient pressure during the firing.

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Table 3. Vacuum chamber test firings

Test	Date	Pressure altitude, ft	Final tipoff angle, rad	Remarks
159	1-8-62	110,000	—	Lost tipoff torque instrumentation
164	1-9-62	125,000	—	Data computation not complete
169	1-12-62	150,000	0.061	
172	1-12-62	110,000	0.007	Mylar seals removed from motor and ignition
174	1-13-62	110,000	0.019	Mylar seals removed
176	1-17-62	150,000	—	Mylar seals removed and excess epoxy trimmed. Data computation not completed
182	1-18-62	110,000	0.026	Mylar seals removed and excess epoxy trimmed

On the premise that the residue of the Mylar seals at the downstream end of the grain and on the igniter were causing disturbance of flow through the primary orifice, these seals were removed for Tests 172, 174, and 176. The starting transient was eliminated. The steady-state spurious torque values, however, were again distributed in a random manner from less than 0.5 ft-lb to about 2.0 ft-lb. To eliminate all possible sources of foreign matter from the motor, the slight excess of epoxy resin, used in inhibiting the grain end and cementing the Mylar seal, was carefully trimmed. The spurious torque characteristics were about the same as for the three previous firings.

On the basis of these test results, it was decided to remove the Mylar seals from the motor to be flown on *Ranger III*. This modification was made at AMR without affecting the flight schedule.

4. Final Checkout of Sphere 011

Subsequent to assembly and acceptance testing of Sphere 011, and prior to shipment, a final functional checkout was performed as planned. The transmitter was found to be inoperative, and the sphere was totally disassembled.

An investigation of the equipment disclosed the presence of an electrical short circuit within the transmitter subassembly. The short circuit occurred between the battery supply lead and circuit ground, represented by a holddown screw on the post amplifier circuit board. Excessive current in the battery lead created heat to burn the wire and the epoxy coating in the immediate vicinity of the wire.

C. Current Programs

1. Spare Capsule

The unused capsule from the *Ranger III* firing will be made available for test purposes, and a verification of the functional characteristics will be completed. These characteristics will be determined (under simulated environments) prior to and at the conclusion of the environmental phases. A life test of the capsule battery will then follow. Upon completion of the tests, the capsule will be disassembled, and the major components tested further.

The RF transmission tests will be of primary interest. Here, the characteristics will be measured and recorded during all of the various phases of the test. The following items will be determined:

- (1) Power output.
- (2) Frequency.
- (3) Spectrum analysis.
- (4) Modulation: variable control oscillator frequency, modulation index, and seismometer operation.

Following initial static testing, a vibration test will be conducted, wherein the lunar capsule will be mounted in a flight configuration dynamic mockup and vibrated to limit loads and levels. Vibration will be introduced in three planes. Sinusoidal and noise inputs will be separately applied. Transmission tests will again be conducted upon completing the vibration test.

The next testing phase will be to impact the lunar capsule sphere on the Hyge machine. The sphere will receive an initial acceleration of approximately 175 g and be impacted at a velocity of 180 ft/sec. Initial acceleration will activate the 25-g switch, which in turn will allow sequential functioning. Activation of the 25-g switch by use of the Hyge machine has already been proved feasible.

The sequence of operation to be determined is as follows:

- (1) Uncaging.
- (2) Postimpact caging.
- (3) Penetrator sphere venting.
- (4) Seismometer centering motor switches.
- (5) Power disconnect switch.

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Following impact, the payload will be X-rayed for determination of caged position and penetrator damage. Transmission tests will again be conducted.

Following the impact test, the payload will be placed in a chamber in which it is possible to simulate a lunar vacuum-thermal environment. Minimum-maximum temperatures of a lunar day will be imposed on the payload for a period of approximately 28 Earth days. Transmission of the payload will be monitored for the entire period to determine the effects of temperature on transmission characteristics. This test will complete the environmental testing. If any battery life remains, the payload will then be placed in an RF quiet-box and operated continuously until failure. Monitoring of relative power level will be conducted daily. A spectrum analysis will be made every 10 days.

The tests outlined will be conducted so long as battery life is available within the payload. Should internal battery failure occur, the payload will be entered, external power applied, and the tests continued to the degree possible. The payload will then be taken apart and a minute examination made of all major components. It is expected that tests will commence in mid-February 1962 and be completed in approximately 9 weeks.

2. Retro-Rocket Nozzle Improvement

Some deterioration of the retro-rocket nozzle near the end of the burning time had been noted. It is clear that, while the deterioration does not degrade performance to a point where flight is inappropriate, an improvement would be desirable. Several courses were examined and after discussion with Hercules Powder Company, a proposal for this work was received.

The retro-rocket nozzle will be fitted with a stainless steel reinforcing cone on the exterior of the nozzle exit cone. This modification is designed to prevent observed geometric deterioration of the aft edge of the cone during burning. Three check firings will be made at full nozzle expansion prior to installing the modification on flight units. These firings will be made into an Aeronutronic-supplied diffuser at the Baccus, Utah, facility of Hercules.

Hercules was authorized to proceed with the design of the modification on January 15, 1962. It is currently expected that the modification will be installed and the three check firings accomplished during the week beginning March 19, 1962. Installation on the flight units will be made at AMR on or before April 1, 1962.

This program is on schedule, and it is probable that the field modifications can be accomplished before flight buildup of the lunar capsule.

3. Spin Motor Improvement

Spin motor testing prior to *Ranger III* indicated desirable areas of improvement, primarily in the further reduction of spurious torques. This program includes incorporation of a number of minor design changes, requalifications of the modified motor, and replacement of the RA-4 and -5 flight motors that were used in pre-*Ranger III* tests.

a. Motor design and fabrication. Design and process modifications included the following items:

- (1) Change in mounting flange installation to improve igniter seal configuration.
- (2) Elimination of the Mylar seal at downstream end of grain.
- (3) Elimination of the 0.050-in. epoxy ring at the grain end.
- (4) Precise control of the exit nozzle contour, both in manufacturing and cold flow adjustment.
- (5) Elimination of fillets in inhibiting and cementing epoxy applications.
- (6) Investigation of a low-pressure propellant that will permit elimination of the primary orifice.

Twenty additional spin motors that include these changes are presently being fabricated. These motors will be utilized as follows:

- (1) Qualification tests (12).
- (2) RA-4 and -5 flights (3).
- (3) Checkout of nozzle trimming tools (2).
- (4) Low pressure propellant tests (2).
- (5) Allowance for loss or spare (1).

Fabrication is proceeding on schedule. The motors for low pressure propellant test and checkout of nozzle trim tools will be available in the week of February 19 to permit commitment of flight configuration. Qualification tests will start about March 1.

b. Igniter testing. Removal of the Mylar seals presents potential problems in long-term exposure of the igniter to expected environmental conditions. A test program is being conducted to verify satisfactory performance of the igniter.

c. Interim testing. Removal of the Mylar seals has eliminated the spurious torque starting transients, but the steady-state values vary in an apparently random manner from about 2 ft-lb to essentially zero. In all cases the exit nozzles are balanced to less than 2 in.-lb of spurious torque by adjustment during cold flow. This adjustment apparently causes small but significant variations in exit nozzle contours that result in random correlation between cold flow and hot firing within the limits experienced.

On new motors, a special expandable honing tool will be used to precisely control the contour. To partially simulate this effect, an adjustment technique was used in three recent motor firings. On these motors, all nozzles were enlarged at least 0.003 in. with the cold flow adjustment tapered abrasive stick, rather than enlarging only the two nozzles required for balance. On all of these motors the steady-state value of spurious torque had a maximum of about 12 in.-lb and averaged less than 6 in.-lb. On the one firing for which data reduction is completed, the tipoff angle is about 0.007 rad.

The other two motors were intended primarily as preliminary grain tests for the low pressure motor. Because of the different burning rate and an absence of the primary orifice, the tipoff data is not directly applicable, but it does furnish additional confirming evidence. On these two motors the higher flame temperature (about 400°F higher than for the present grain) caused exit nozzle erosion of about 0.003 to 0.004 in. (diameter) as compared to about 0.0005 to 0.001 in. in the present design. As expected, this caused a relatively steadily increasing torque unbalance as the run progressed, with highest values at the end of the run.

d. Capsule separation gas dynamics. The separation test fixture was completed on schedule and installed in the vacuum chamber at Douglas, Long Beach, on January 8, 1962. The installation in the chamber is shown in Figure 2. This figure shows the bottom of the bus simulator, the bus carriage, and the main frame. For steady-state cold flow tests, the relative position of the capsule and bus is adjusted in rotation by the motor and gear reducer attached to the bus mounting shaft, and in separation by the motor-driven lead screw at the right-hand side of the main frame. For dynamic tests, the separation acceleration is provided by dead weight. The bus can be rotated during dynamic separation tests, but the angular rate (about 0.5 rad/sec) does not afford accurate simulation.

The upper surface of the bus simulator and retro-motor support are shown in Figure 3. The load sensitive portion

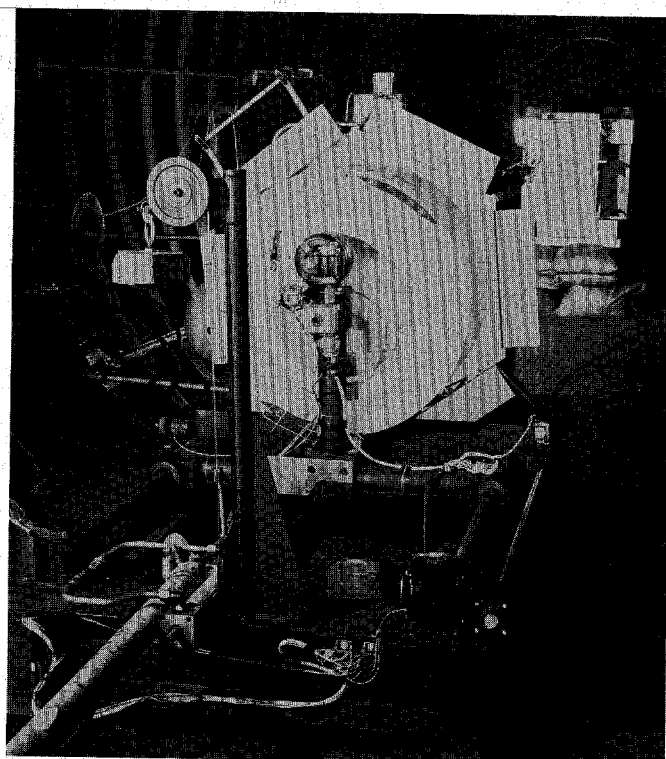


Figure 2. Separation fixture in vacuum chamber

consists of the nozzle and the motor aft dome and mounting flange.

Because of the urgency connected with spin motor firings on the torque fixture prior to *Ranger III*, the separation fixture tests were postponed. Some cold flow tests have now been conducted, and data analysis is underway. Definitive results are not yet available. No motor firings have been made on this test fixture.

e. Spin restraint design. As backup for the present separation system, the design of a spinup restraint that could be added to the present system is being pursued. In concept, the device will hold the capsule centered at the plane of the retro-motor exit nozzle for the first portion of the spinup, until the capsule roll rate affords gyroscopic stability. The restraint will be in the lateral direction only, allowing unrestricted spinup and separation and will be effective for the first few inches of separation only. A tradeoff study is now being made to determine the most feasible compromise between restraint length, restraint effectiveness, and design complexity. Obvious design problems being studied include furnishing the required rigidity during spinup and flexibility during boost, and injection vibration.

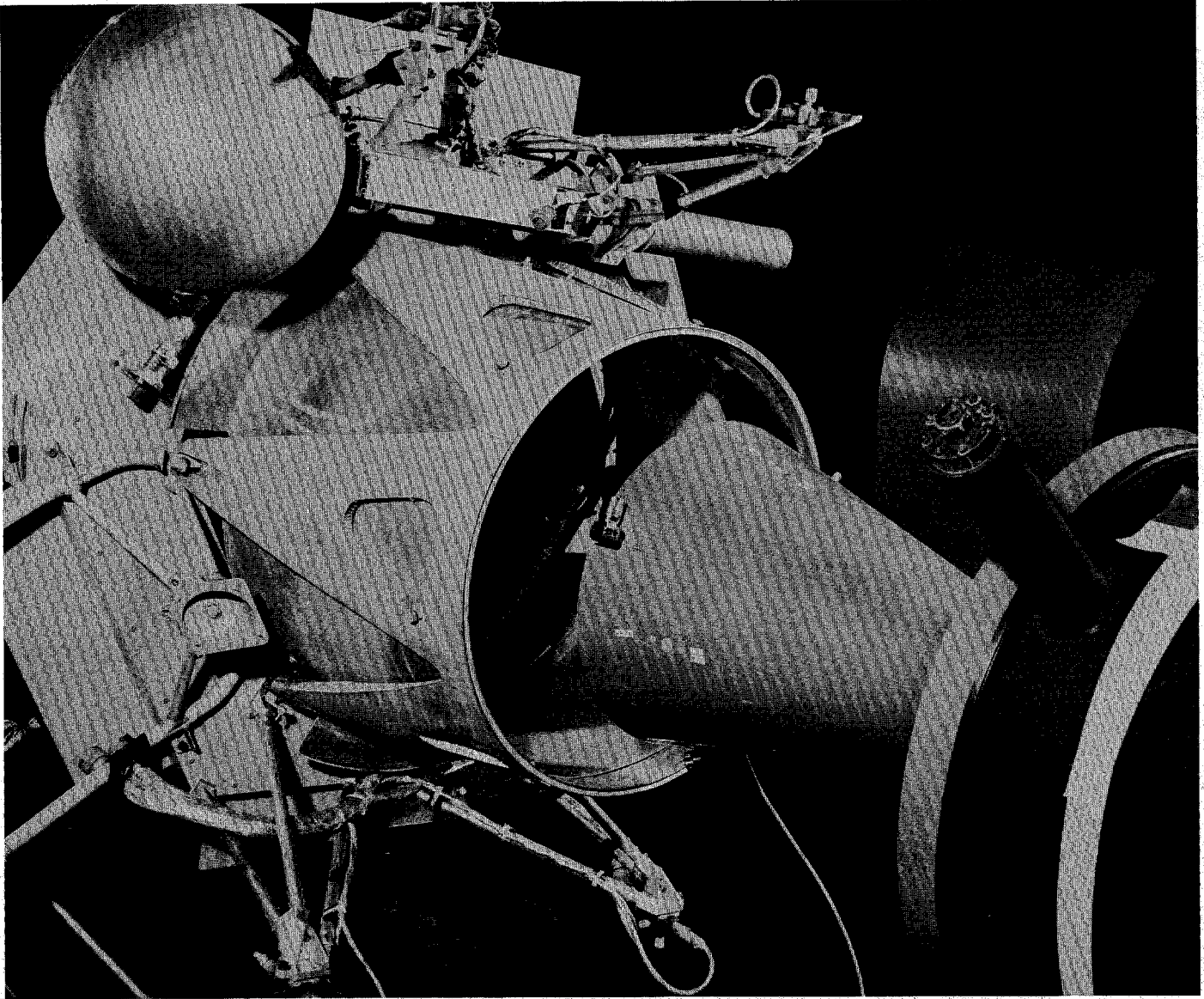


Figure 3. Bus and support structure simulator

4. Ranger RA-4

a. Assembly status. The assembly schedule for *Ranger III* called for delivery of two lunar capsules in the first week of January 1962. Payload spheres Serial No. 010 and 011 had been assembled to meet this schedule. On the shipping date it was discovered, as the result of an RF test, that Serial No. 010 was no longer operative. Assembly of Serial No. 012 was then initiated. Time, however, did not permit a sterile assembly and this requirement was waived. Under these conditions Serial No. 012 was assembled and delivered to the Atlantic Missile Range on Jan-

uary 17, 1962. Serial No. 012 was subsequently used as the *Ranger III* payload.

Preparatory subassemblies are being made for RA-4 payloads Serial Nos. 013 and 014. Actual assembly will start on February 19 for 013, and February 26 for 014. It is expected that both payloads will be shipped on April 1, 1962.

In mid-April 1962 the assembly of Serial No. 015 will start in the event that there is a postponement of the April firing.

b. Weights and performance. The weight summary for *Ranger RA-4* is not clearly defined at this time. Minor uncertainties exist as a result of product improvement in some areas such as the retro-motor nozzle. It is estimated that the total capsule weight will be 332.1 lb, an increase of 1.9 lb over *Ranger III*. About 1.5 lb of increased weight will result from increased payload to compensate for the lower impact velocity of the *RA-4* trajectory. Table 4 is a breakdown of the predicted *RA-4* weights compared to the *Ranger III* weights. The estimated weight, center-of-gravity, and inertia of the system are shown in Table 5.

Table 5. Operational schedule weight, center-of-gravity, and inertia

Configuration	Weight, lb	Center-of-gravity (z), in.	Inertia, slug-ft ²		
			<i>I</i> _{roll}	<i>I</i> _{pitch}	<i>I</i> _{yaw}
Launch	332.1	449.4	3.47	11.89	11.67
After separation	310.2	448.2	2.58	9.43	9.43
Burnout	113.4	437.7	0.89	3.51	3.51
Impact	89.3	432.7	0.68	0.66	0.67

Table 4. Lunar capsule weight summary

Component	Predicted RA-4 weight, lb	Ranger III weight, lb	Component	Predicted RA-4 weight, lb	Ranger III weight, lb
Payload			Motors		
Survival sphere electronics, antenna, batteries, wiring	23.8	24.0	Retro-rocket motor and igniter	214.4	214.4
Structure, insulation, devices	13.7	13.5	Spin motor, igniter, and attachments	2.3	2.4
Water	3.6	3.6	Total separated weight	310.2	308.8
Seismometer	7.8	7.8	Bus mounted equipment		
Flotation fluid and outer shell	8.2	8.2	Altimeter and antenna	6.6	6.6
Total survival sphere	57.1	57.1	Altimeter support and deployment	2.2	2.1
Impact limiter	32.2	32.2	Motor support structure and separation	3.8	3.8
Vibration dampers and clamp	1.0	1.0	J-box and connectors	0.8	1.0
Balance weights (motor)	0.4	0.3	Radiation shield	5.0	3.5
Power sequence unit, wiring, and motor thermal shield	1.3	1.4	Spin motor vent	3.5	4.4
Ballast	1.5	0	Total lunar capsule	332.1	330.2
Total payload	93.5	92.0			

III. Ranger TV Subsystem

A high resolution TV experiment is required for support of the manned lunar program. RCA under contract to JPL is developing a TV subsystem for early integration into the *Ranger* Project; *Rangers* RA-6, -7, -8, and -9 have been selected for the mission. The RCA TV subsystem will mate with the JPL-produced bus which will remain essentially unchanged from the lunar capsule series.

The mechanical test model (MTM) has been assembled and tested at RCA, and shipped to JPL for further tests. The thermal test model (TTM) assembly has been started, and engineering models on all subassemblies have been received and evaluated. Design changes necessary for proper operation have been initiated, to be implemented in the qualification test models and proof test models. The first GSE equipment has been delivered and installed at the Astro-Electronics Division (AED), with fabrication and assembly of the second GSE nearly complete.

A. Basic Structure

Efforts to reduce the weight of the structure have resulted in several design changes. The cylindrical hat section which was previously subjected to the relatively heavy camera loads was redesigned to carry only the light

omni-antenna loads; a weight saving of approximately 6 lb was realized. Other weight reduction changes involved thermal shroud-central box gussets, and beams in the base section.

The conical shroud is now fabricated in two halves which are mechanically fastened to the main structure with removable fasteners. This simplifies equipment mounting and provides full and ready access to the interior. It is anticipated that, as a part of the removable fastener unit, Davis press-fit locknuts will be used in place of Rivnuts primarily because of their greater reuse capacity, as well as their greater reliability.

The thermal conical shroud is cut out locally in five areas to ensure the clear line of vision required for the camera. The cutout for the sixth camera is provided in the shroud of the cylindrical hat section. The cutouts in the conical shroud are flanged internally to provide a more rigid periphery.

On January 27, the MTM successfully met the vibration test requirements and the unit was shipped to JPL. After completion of the vibration-mode survey tests the test data will be analyzed to determine the degree of correlation with the theoretical results.

Strain gage and accelerometer data recorded during qualification testing at AED will be analyzed to evaluate amplification factors and load distribution in the structure; the degree of correlation with theoretical results will be established.

1. Stress Analysis

This is a preliminary stress analysis to verify the integrity of the structural design for the *Ranger* TV subsystem; a detailed analysis has been initiated. Table 1 summarizes the loading criteria used in the design and analysis of the basic structure.

a. Bending stresses in the rectangular box section. At the base of the structure, the base moment may be calculated from the following equation:

$$M = Wg\bar{z}$$

where M = the base moment, W = the structure weight, g = the lateral loading, and $\bar{z}$ = the moment arm. The weight of the payload = 346.7 lb, and the moment arm = 17 in.; therefore for a 15.6-g lateral loading, the base moment = 91,900 in.-lb.

The section modulus may be determined from the following equation:

$$S = \frac{I_{xx}}{c}$$

where I_{xx} = the moment of inertia of the box about the y - x axis, and c = the distance from the centroid to the extreme fibre of the box. For an I_{xx} of 59.3 in.⁴ and a c of 4.655 in., $S = 12.74$ in.³

The bending stress (F_b) is equal to the base moment (M) divided by the section modulus (S); therefore, for an M of 91,900 in.-lb and an S of 12.74 in.³, F_b is equal to 7220 psi.

The crippling stress (F_{cc}) may be calculated from the equation,

$$F_{cc} = KE(t/b)^2$$

where K is the buckling or crippling constant = 3.8 for $a/b = 7.25/4.11 = 1.76$ (a is defined as the height between bulkheads), E is the modulus of elasticity = 10^7 psi, $t/b = 0.062/4.11 = 0.0151$ (t is the thickness of the skin of plate, and b is the width between supports). Therefore, based on the above equation and values, F_{cc} is equal to 8650 psi. This allowable stress is based on the size of the long panel with three equally-spaced stiffeners, all sides being simply supported.

The margin of safety (MS) is equal to $F_{cc}/F_b - 1$; therefore, from an F_{cc} of 8650 psi and an F_b of 7220 psi, $MS = 0.20$.

b. Spacecraft attachment loads. The mounting arrangement for the structure is shown in Figure 1. The conditions for loading may be summarized as:

- (1) The weight = 346.7 lb.
- (2) The cg is located ≈ 22 in. above the foot of the structure base.
- (3) The lateral loading = 15.6 g.
- (4) The longitudinal loading = 20.8 g.

The vertical-mount reactions (summarized in Table 2) are the result of the following calculations:

Bathtub mount reactions with a 15.6-g lateral load about the y-y axis

$$F_{1z} = -F_{4z} = 4F_{2z}, F_{2z} = F_{6z} = -F_{3z} = -F_{5z}$$

$$M = F_{1z}(24) + F_{2z}(12) + F_{6z}(12)$$

$$= F_{1z}(24) + F_{2z}(24)$$

$$= F_{1z}(24) + 6F_{1z}$$

$$= 30F_{1z}$$

$$F_{1z} = -F_{4z} = \frac{346.7 \times 22 \times 15.6}{30} = 3970 \text{ lb}$$

$$F_{2z} = F_{6z} = -F_{3z} = -F_{5z} = \frac{3970}{4} = 992 \text{ lb}$$

20.8-g longitudinal load

$$F_{1z} = F_{2z} = F_{3z} = F_{4z} = F_{5z} = F_{6z}$$

$$= \frac{346.7 \times 20.8}{6} = 1202 \text{ lb}$$

15.6-g lateral load about the x-x axis

$$\therefore F_{2z} = F_{3z} = -F_{5z} = -F_{6z} = \frac{346.7 \times 22 \times 15.6}{2 \times 20.76} = 2875 \text{ lb}$$

c. Thermal shroud analysis. Figure 2 illustrates the *Ranger* TV payload thermal shroud. Assuming that the shroud sustains the total load, the moment about the base is the shroud weight (346.7 lb) times the moment arm (17 in.) or $M = 346.7 \times 17 = 5880$ in.-lb/g. Under a lateral loading of 15.6 g, the total bending moment is 91,900 in.-lb.

The bending stress is determined by dividing the product of the bending moment and the radius (r) by the structure's moment-of-inertia. For a moment of 91,900 in.-lb, a radius of 12.45 in., and a thickness of 0.032 in.,

$$F_b = \frac{91,900 \times 12.45}{12.45 \times 0.032\pi} = 5875 \text{ psi}$$

Table 1. Design criteria for TV subsystem structure

Parameters	Load, g					
	Static		Shock		Vibration	
	Limit	Ultimate design ^a	Limit ^b	Ultimate design ^a	Limit	Ultimate design ^a
Longitudinal	10	12.5	18.4	23.0	16.6 ^c	20.8
Lateral	2	2.5	3.54	4.43	12.5 ^d	15.6
Type of allowable stress ^e	F_{TY}	F_{TU} or F_{CC}	—	F_{TU} or F_{CC}	—	F_{fat} or F_{CC}

^aUltimate design load: 1.25 limit load.
^bIncludes effect of shock-wave form and duration, and the natural frequency of the system.
^cIncludes amplifications due to white-Gaussian noise.
^dIncludes amplification factor of 10.
^e F_{TY} = tensile yield strength; F_{TU} = tensile ultimate strength; F_{CC} = crippling strength; and F_{fat} = fatigue allowable stress (includes allowance stress concentration).
 NOTE: For 24ST-4, $F_{fat} = \pm 17,700$ psi for lateral loads (accounts for cumulative effect of lateral and longitudinal cycling).

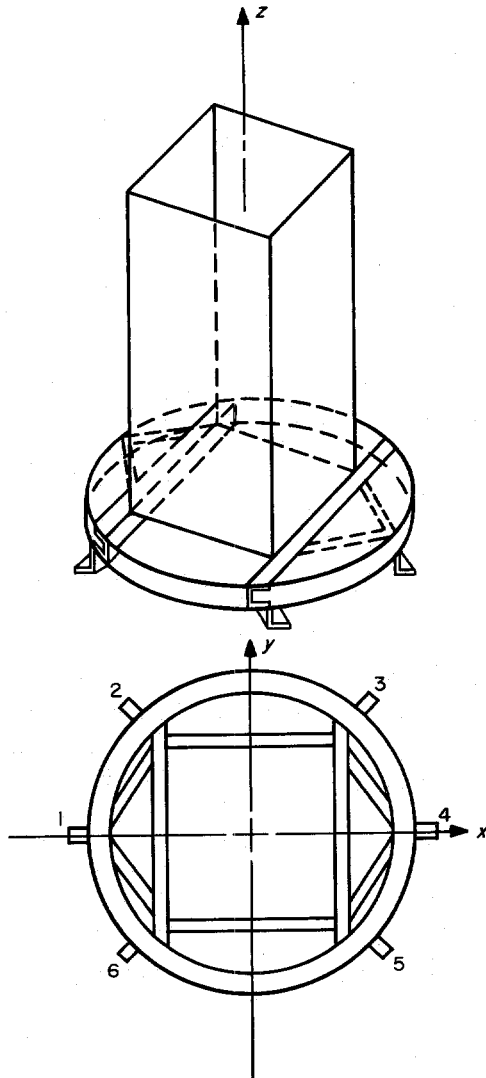


Figure 1. Mounting arrangement

Table 2. Vertical-mount reactions, lb

Load, g	Condition	Reactions, lb					
		F_{1z}	F_{2z}	F_{3z}	F_{4z}	F_{5z}	F_{6z}
15.6	Lateral y-y axis	3970	992	-992	-3970	-992	992
20.8	Longitudinal	1202	1202	1202	1202	1202	1202
15.6	Lateral x-x axis	0	2875	2875	0	-2875	-2875

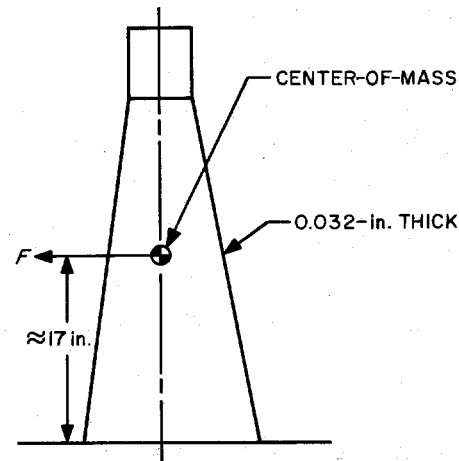


Figure 2. Thermal shroud

The stress (F_s) at the shroud base is determined by dividing the force by the area of material. The force is the shroud weight multiplied by the lateral g -loading, and the area is given by $2\pi rt$. F_s is found to be 2160 psi, and the maximum allowable stress is simply twice F_s .

From previous calculation, the allowable bending stress based on the fatigue strength is 17,700 psi. The shroud Margin of Safety (MS) in bending is found to be 2.01.

Consideration must also be given to the buckling compressive stress in the curved panels of the thermal shroud. Referring to Reference 1, the buckling compressive stress (F_{CCR}) for the shroud structure is given by the following equation:

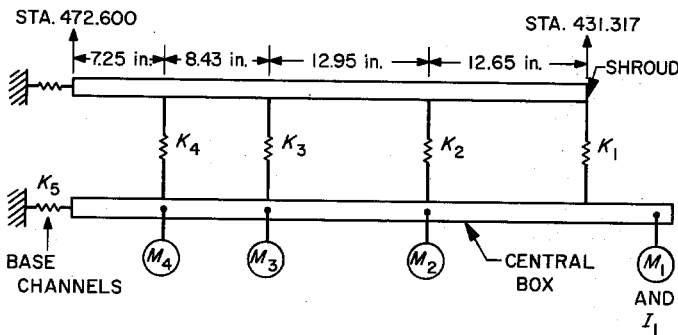
$$F_{CCR} = 0.45E (t/r)$$

where

E = Modulus of elasticity (10.6×10^6 psi)

$$t/r = 0.032/12.45 = 0.00257$$

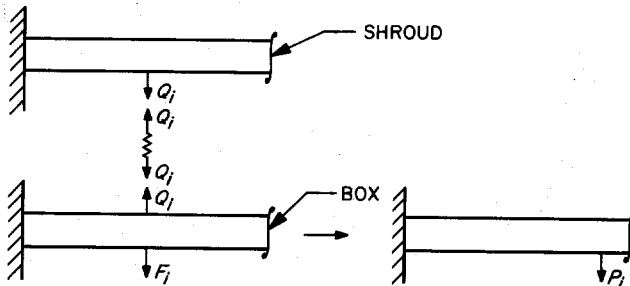
The constant in the equation, 0.45, is determined by the shape of the shroud. Substituting the values in the equation, F_{CCR} is found to be 12,270 psi; the shroud MS is 1.085. Based on the calculations outlined, it can be seen that the shroud will successfully withstand the full lateral g -loading in compressive buckling.



WHERE

M_1 = MASS OF CAMERAS AND ASSOCIATED STRUCTURE
 M_2 , M_3 AND M_4 = EQUIVALENT MASSES OF COMPONENT BOXES AND STRUCTURE
 K_1 = SPRING RATE OF RINGS CONNECTING SHROUD TO CENTRAL BOX, lb/in.
 K_2 , K_3 AND K_4 = SPRING RATE OF BULKHEADS FOR IN-PLANE LOADS, lb/in.
 K_5 = SPRING RATE OF BASE CHANNEL BEAMS

Figure 3. Assumed vibration system



WHERE

F_i = INERTIA FORCES DUE TO m_i
 Q_i = SPRING FORCE, APPLIED SHROUD FORCE
 P_i = APPLIED BOX BEAM FORCE
 $P_i = F_i - Q_i$

Figure 4. Force symbolism used in interaction equations

2. Vibration Analysis

The idealized four-mass system assumed for purposes of lateral vibration analysis is shown in Figure 3. The system flexibility includes that of the thermal shroud, central box, the bulkheads between the thermal shroud and box, and the channel beams in the base section. The effect of variable flexibility in the box and shroud are accounted for. Shear as well as bending deflections are calculated to determine spring rates.

The equations of motion for the four masses are written with a consideration of the inertia forces acting during free vibration of the system. A detailed procedure for this analysis is discussed in the following. The force symbolism used in the interaction equations is shown in

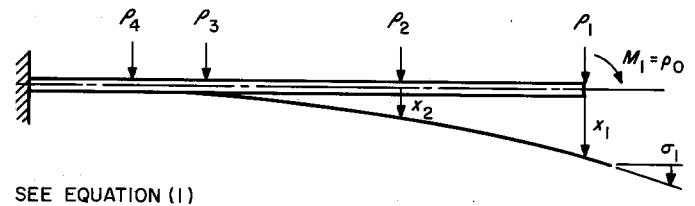


Figure 5. Control box beam free body diagram

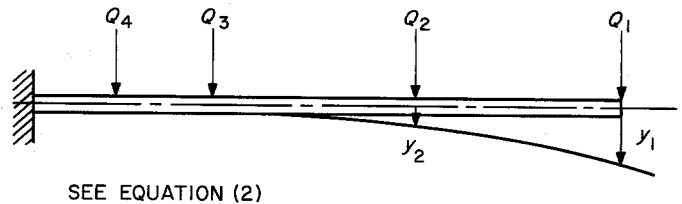
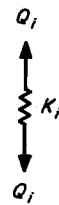


Figure 6. Thermal shroud free body diagram



SEE EQUATION (3)

WHERE $i = 1, 2, 3, 4$

Figure 7. Spring free body diagram

Figure 4, the control box beam free body diagram and deflection equations in Figure 5, the thermal shroud free body diagram and deflection equations in Figure 6, and the spring free body diagram and deflection equations in Figure 7.

$$\begin{aligned} O_1 &= \alpha_{00}\rho_0 + \alpha_{01}\rho_1 + \alpha_{02}\rho_2 + \alpha_{03}\rho_3 + \alpha_{04}\rho_4 \\ X_1 &= \alpha_{10}\rho_0 + \alpha_{11}\rho_1 + \alpha_{12}\rho_2 + \alpha_{13}\rho_3 + \alpha_{14}\rho_4 \\ X_2 &= \alpha_{20}\rho_0 + \alpha_{21}\rho_1 + \alpha_{22}\rho_2 + \alpha_{23}\rho_3 + \alpha_{24}\rho_4 \\ X_3 &= \alpha_{30}\rho_0 + \alpha_{31}\rho_1 + \alpha_{32}\rho_2 + \alpha_{33}\rho_3 + \alpha_{34}\rho_4 \\ X_4 &= \alpha_{40}\rho_0 + \alpha_{41}\rho_1 + \alpha_{42}\rho_2 + \alpha_{43}\rho_3 + \alpha_{44}\rho_4 \end{aligned} \quad (1)$$

where α_{ij} is the deflection of ρ_i due to unit load applied at the location of ρ_j

$$\begin{aligned} Y_1 &= \beta_{11}Q_1 + \beta_{12}Q_2 + \beta_{13}Q_3 + \beta_{14}Q_4 \\ Y_2 &= \beta_{21}Q_1 + \beta_{22}Q_2 + \beta_{23}Q_3 + \beta_{24}Q_4 \\ Y_3 &= \beta_{31}Q_1 + \beta_{32}Q_2 + \beta_{33}Q_3 + \beta_{34}Q_4 \\ Y_4 &= \beta_{41}Q_1 + \beta_{42}Q_2 + \beta_{43}Q_3 + \beta_{44}Q_4 \end{aligned} \quad (2)$$

$$\begin{aligned}
 \delta_1 &= \frac{Q_1}{K_1} \\
 \delta_2 &= \frac{Q_2}{K_2} \\
 \delta_3 &= \frac{Q_3}{K_3} \\
 \delta_4 &= \frac{Q_4}{K_4}
 \end{aligned}
 \quad (3)$$

Geometrical equations

$$\begin{aligned}
 \chi_1 &= Y_1 + \delta_1 \\
 \chi_2 &= Y_2 + \delta_2 \\
 \chi_4 &= Y_4 + \delta_4
 \end{aligned}
 \quad (4)$$

Inertia force equations

$$\begin{aligned}
 F_0 &= IO_1\omega^2 \\
 F_1 &= m_1\chi_1\omega^2 \\
 F_2 &= m_2\chi_2\omega^2 \\
 F_3 &= m_3\chi_3\omega^2 \\
 F_4 &= m_4\chi_4\omega^2
 \end{aligned}
 \quad (5)$$

also

$$\begin{aligned}
 F_0 &= \rho_0 \\
 F_1 &= \rho_1 + Q_1 \\
 F_2 &= \rho_2 + Q_2 \\
 F_3 &= \rho_3 + Q_3 \\
 F_4 &= \rho_4 + Q_4
 \end{aligned}
 \quad (6)$$

Relationships between the ρ and Q loads are found by substituting the equations in Sets (1), (2), and (3) into their corresponding equations in Set (4). In turn, these relationships and the equations in Set (5), are substituted into Set (6). The resulting expressions relate to the ρ and inertia loads. The final deflection equation used to obtain the natural frequencies are obtained by substituting the expressions obtained in the preceding step into equation Set (1). The coefficient determinant of the final set of equations is then set equal to zero; a fifth order equation, corresponding to the number of degrees of freedom of the system, is obtained. The natural frequencies of the system are then readily determined from the roots of this equation.

B. Cameras

The first partial scan (Type-P) camera electronics package of the engineering model has been delivered and placed in operation, using a camera-shutter combination fabricated at AED. Immediate plans include a series of comprehensive measurements on all phases of camera operation to establish criteria for acceptable equipment performance under all types of environment and to establish acceptance requirements of future equipments.

Vidicon evaluation and test has continued. Consultation between RCA and JPL regarding proposed reticles for the Type-F and -P cameras has resulted in a redesign of both (Figs 8 and 9).

1. Optics

No manufacturing problems were experienced for the six prototypes or the first nine production 3 in. $f/2.0$ super Baltar lenses. The prototypes were photographed, tested on a lens bench, and results recorded (Tables 3 and 4). Photographic resolution film strips were sent to RCA for evaluation. All prototypes satisfied optical specifications.

Prototype 759-2 was returned to Bausch and Lomb after completion of environmental testing. It has been rephotographed, and indications are that it is as good as when first photographed. Bench measurements will also be made for comparison with pre-environmental test data. The lens will then be returned to RCA for disposition.

Bausch and Lomb also conducted a test to determine compatibility of magnesium fluoride, anti-reflection coating, and ethylene oxide gas. Contact optics were exposed to a sterilization atmosphere. For this experiment, the optical surfaces must be clean and absolutely free of

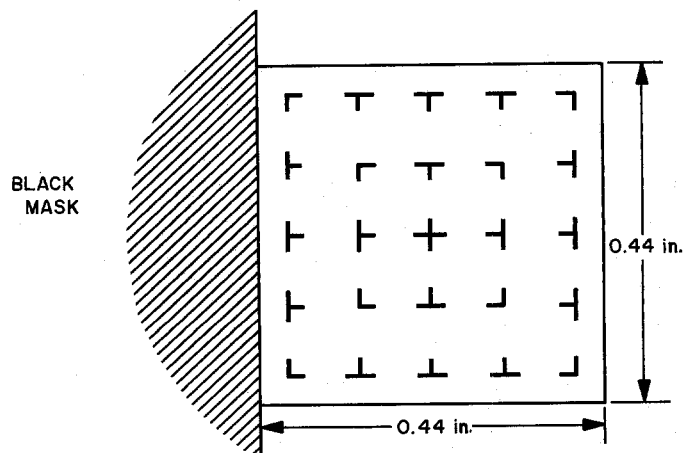


Figure 8. Vidicon reticle for Type-F cameras



Table 3. Lens resolution test data

NOTE: USAF bar target photographed on Kodak high-contrast copy film, developed in D-19 for 6 min at 68°F with constant agitation.

finger marks and smudges. No deleterious effects were noted to either the coating or the edge lacquers.

The data in Tables 3 and 4 is that taken at Bausch and Lomb on the environmental test model lenses and is of interest in evaluating the performance of the TV subsystem cameras.

2. Analysis of the Color Camera

A system has been proposed, based on the use of a three-gun kinescope, for display and photography of the Camera A color pictures of the Moon's surface. Subse-

quently, it was decided that this system could be simplified by sequentially projecting the three Camera A color images directly onto sensitized printing paper. This method eliminates the equipment required to convert the sequential color camera signals to the simultaneous electrical signals necessary for the three-gun kinescope.

In order to design a color experiment capable of accurately reproducing the color in the original scene, the following must be determined:

- (1) Specifications of the light source which will project color images onto the printing paper.

Table 4. Lens general test data

Lens	Radial distortion at 6 deg, %	Relative illumination at 6 deg, %	Transmission of lens at $f/2.0$, %	EFL ^a , mm	BF ^b , mm
FF153	0.098	89	90	74.70	46.30
749-1	0.026	86.3	91	74.70	46.35
749-2	0.026	86.5	89	46.40	74.60
749-3	0.051	87	90.5	74.85	46.45
749-4	0.051	86.3	92	74.60	46.15
749-5	0.051	86.5	92	74.55	46.35
749-6	0.090	87	92	74.60	46.40

^aEFL: equivalent focal length.
^bBF: back focal.

Table 5. Primary sources

Source	Chromaticity coordinates			Relative values of primaries required to reproduce the color of sunlight
	x_0	y_0	z_0	
Tungsten (3200°K) + interference filter	0.137	0.059	0.804	1.00
Tungsten + Wratten 58	0.261	0.689	0.051	0.77
Tungsten + Wratten 25	0.685	0.315	0.000	—
Direct sunlight	0.318	0.329	0.353	0.61

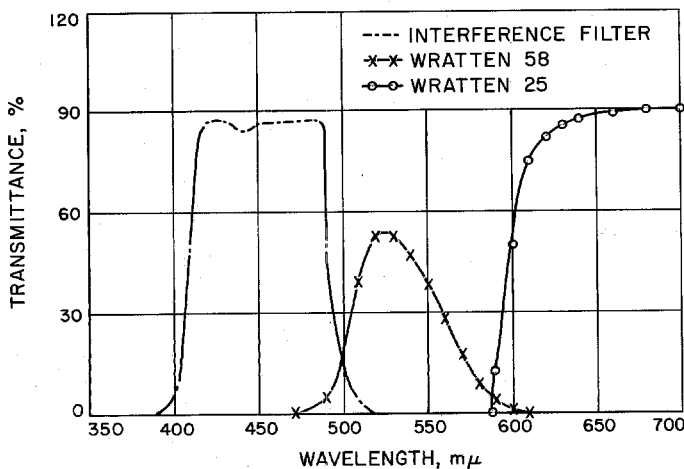


Figure 10. Transmission range of blue, green, and red filters

- (2) Required spectral response of the *Ranger* vidicon.
- (3) Transmittance of the optical filters.
- (4) Accuracy of the color reproduction.
- (5) Illumination required for adequate signal-to-noise ratio (and high quality pictures).

a. Primary source. In order to obtain color prints, a Chromega enlarger (having a tungsten light source operating at a color temperature of 3200°K) will be used in conjunction with three filters having the transmittances shown in Figure 10. The chromaticities of the light from the tungsten lamp after passing through the individual filters are listed in Table 5 and are plotted on a chromaticity diagram (Fig 11). The triangle formed by the three primaries represents the colors which can be formed by additive mixture of the three. Colors located outside this triangle will be reproduced having chromaticities represented by points located on the perimeter of this triangle.

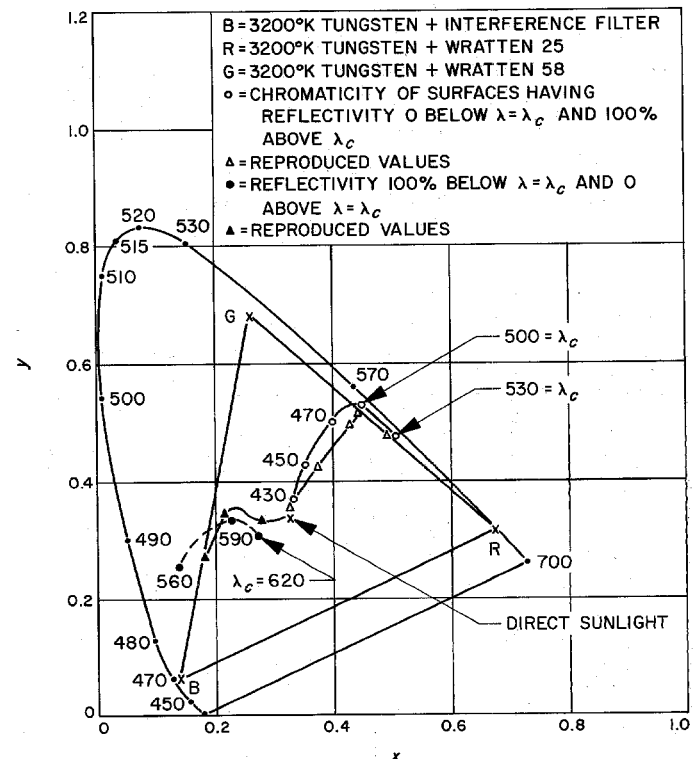


Figure 11. Chromaticity diagram of experimental light sources

In order to form a color image of the original scene, the relative amounts of each primary required to reproduce the color of a test object must be determined. These values have been calculated based on the assumption that sunlight reflected from neutral objects (reflectivity independent of wavelength) must be correctly reproduced. The chromaticity of direct sunlight as calculated from the spectral data given in Reference 2, and the relative amounts of the primary source required to reproduce this chromaticity, are listed in Table 5.

b. Determination of the theoretical response of the vidicon. Since the color camera controls the amounts of the primaries, its spectral response must be determined.

This has been done according to the method outlined in the original proposal, resulting in the following equations:

$$\bar{r} = 0.733\bar{X} - 0.271\bar{Y} - 0.105\bar{Z}$$

$$\bar{g} = -0.267\bar{X} + 0.578\bar{Y} + 0.0033\bar{Z}$$

$$\bar{b} = 0.0129\bar{X} - 0.0279\bar{Y} + 0.316\bar{Z}$$

where $\bar{X}$, $\bar{Y}$, and $\bar{Z}$ are the CIE tri-stimulus values of the spectrum, and r , b , and g are the required spectral responses of the vidicon to red, blue, and green (Fig 12).

The curves of the theoretical response in Figure 12 have both negative and positive values. Since negative responses are not physically realizable, modified curves have been drawn showing response in positive values only. The loss of the negative areas is partially offset by adjusting the positive areas so that the areas under the new curves equal the areas under the original ones.

c. Optical filters. The same vidicon is used for each color on a time-sharing basis. Its input, therefore, must be modified by optical filters, for sequential separation of the three primary colors in the scene being imaged. The theoretical response of the vidicon is shown in Curve B of Figure 13. Curve A is the actual response for a vidicon from the group being produced for the *Ranger* program. Using Curve A, the required transmittances of the optical filters have been calculated for the positive camera responses. These values are shown in Figure 14.

d. Accuracy of color reproduction. Since the negative values are not included in the camera spectral responses, all colors cannot be accurately reproduced.

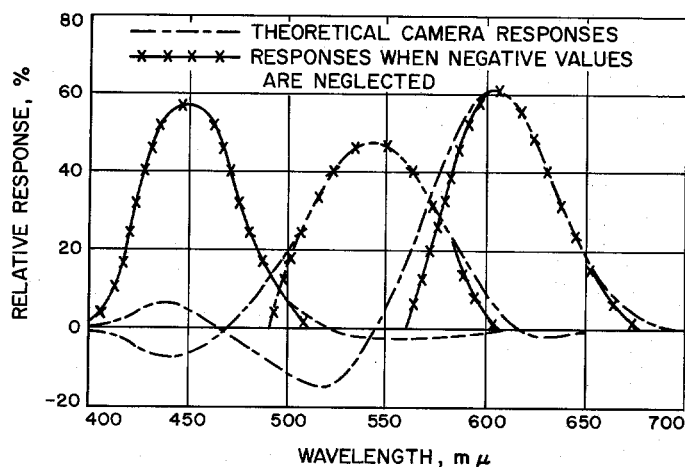


Figure 12. Theoretical spectral response of *Ranger* vidicon

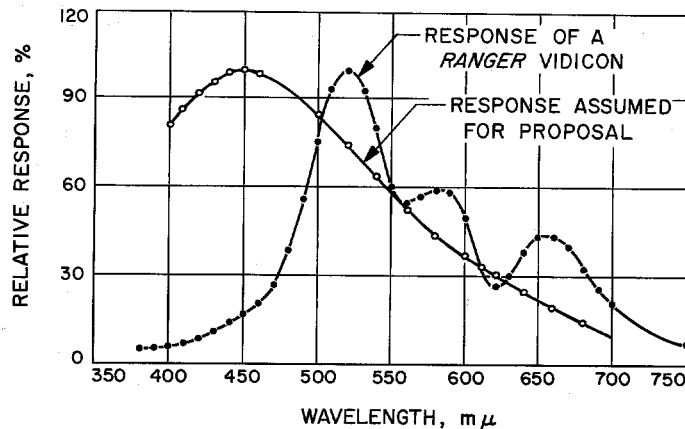


Figure 13. *Ranger* vidicon spectral response (with ASOS photoconductor)

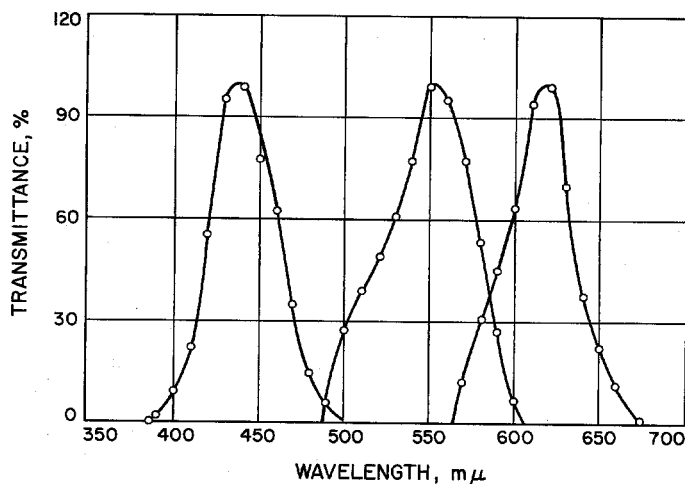


Figure 14. Transmission range of filters

The ability of a system consisting of the vidicon (Fig 13, Curve A) and the optical filters (Fig 14) to reproduce the color of surfaces having hypothetical reflectivities has been determined and is shown in Figure 11. The spectral reflectivities of these surfaces are either 100% or zero above or below a wavelength λ_c .

Figure 11 shows that all reproduced colors are within the triangle formed by the primary sources. The chromaticity differences between the actual and reproduced values can be compared to the standard deviation recorded in the determination of just-noticeable chromaticity differences. This has not been done, since the actual transmittance of a set of optical filters may be different from the calculated values shown in Figure 14.

e. Illumination requirements. Using the filter transmittances of Figure 14 and the spectral sensitivity of the *Ranger* vidicon (Fig 13, Curve A), the response of

Camera A to blue, green, and red has been calculated. The relative values of these responses with respect to that of a vidicon viewing the scene directly (without filters) are given in Table 6.

Since the vidicon's transfer characteristic (or gamma) cannot be changed as the appropriate filters are placed between the vidicon and the scene, the signal-to-noise ratio will vary as shown in Figure 15. From this figure it can be seen that the quality of the blue image is poor even under maximum illumination for black and white, making the color experiment marginal.

f. Performance limitations. Table 6 shows that the response of the *Ranger* vidicon to blue is approximately one-fifth of that specified in the design proposal. This reduction in response is due to the fact that the spectral sensitivity of the vidicon is much lower in the blue region of the spectrum than that specified. Since the major emphasis is placed on obtaining black and white images of the Moon's surface, this spectral sensitivity is ideal, due to the fact that it approximates eye sensitivity, a condition which is necessary if the luminances of colored objects are to be reproduced in appropriate shades of gray.

Table 6. Camera responses

Vidicon	Primary color				
	Blue	Green	Red	Black and white	
				No Filter ^a	Filter ^b
Ranger vidicon	0.059	0.35	0.14	1.0	—
Vidicon specified in design proposal	0.32	0.23	0.123	1.0	0.32

^aNo filter is required with the *Ranger* vidicon since its spectral curve approximates the eye sensitivity.

^bThis filter makes the camera spectral sensitivity equal the eye response.

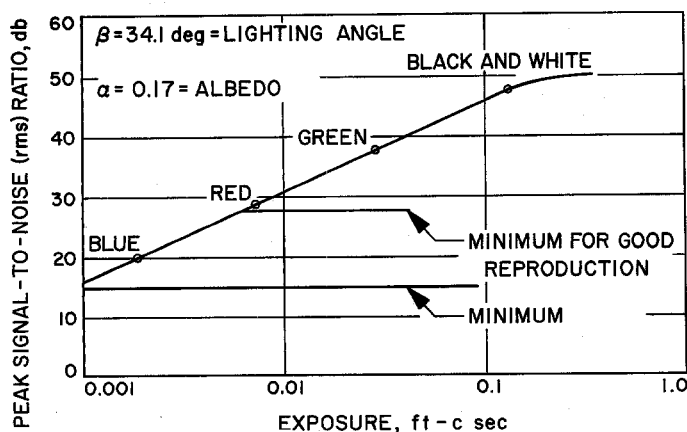


Figure 15. *Ranger* vidicon transfer characteristics

This fact will restrict the lighting conditions for which the full color capability will be available. However, the color subsystem design is proceeding on this basis. Under favorable lighting the color will be obtained. When the lighting conditions for a flight are marginal, the black and white pictures will still be obtained on a non-interference basis. The resolution from frame to frame does not change rapidly and the exposures for color do not detract from the sampling for black and white pictures.

C. Control Programmer and Camera Sequencer

The camera sequencer design has been completed and the engineering model has been received at AED from RCA, Burlington. Electrical acceptance tests have been performed, and bench tests conducted with the sequencer connected to a load box and temperature varied from -10 to 65°C . Figure 16 shows the modules which comprise the control programmer and camera sequencer, the sequencer power supply, and the video combiner.

The engineering model of the analog-to-digital converter has been fabricated and is being installed in the sequencer for tests with the color wheel. Figure 17 shows the engineering model of the color wheel which is designed as an ancillary to the Type-F cameras and is normally attached to Camera A. Color filters will be fitted into the circular cup holds. Operation of the filter wheel control system was found to be essentially satisfactory. Advancing of each quadrant was accomplished in less than 1 sec by the gear-reduction drive.

A series of temperature tests were performed on the digital module section (A9) of the camera sequencer. All input-output signals were satisfactory except that four partial-scan erase signals had $1\text{-}\mu\text{sec}$ transients one or two times during the erase period. A circuit change has been developed, and is undergoing environmental temperature testing to enable incorporation if required. (All other camera sequencer functions were satisfactory in both high and low temperature environments.)

The video combiner engineering model was integrated with the camera sequencer and temperature tested. Each of the channels indicated satisfactory performance. The

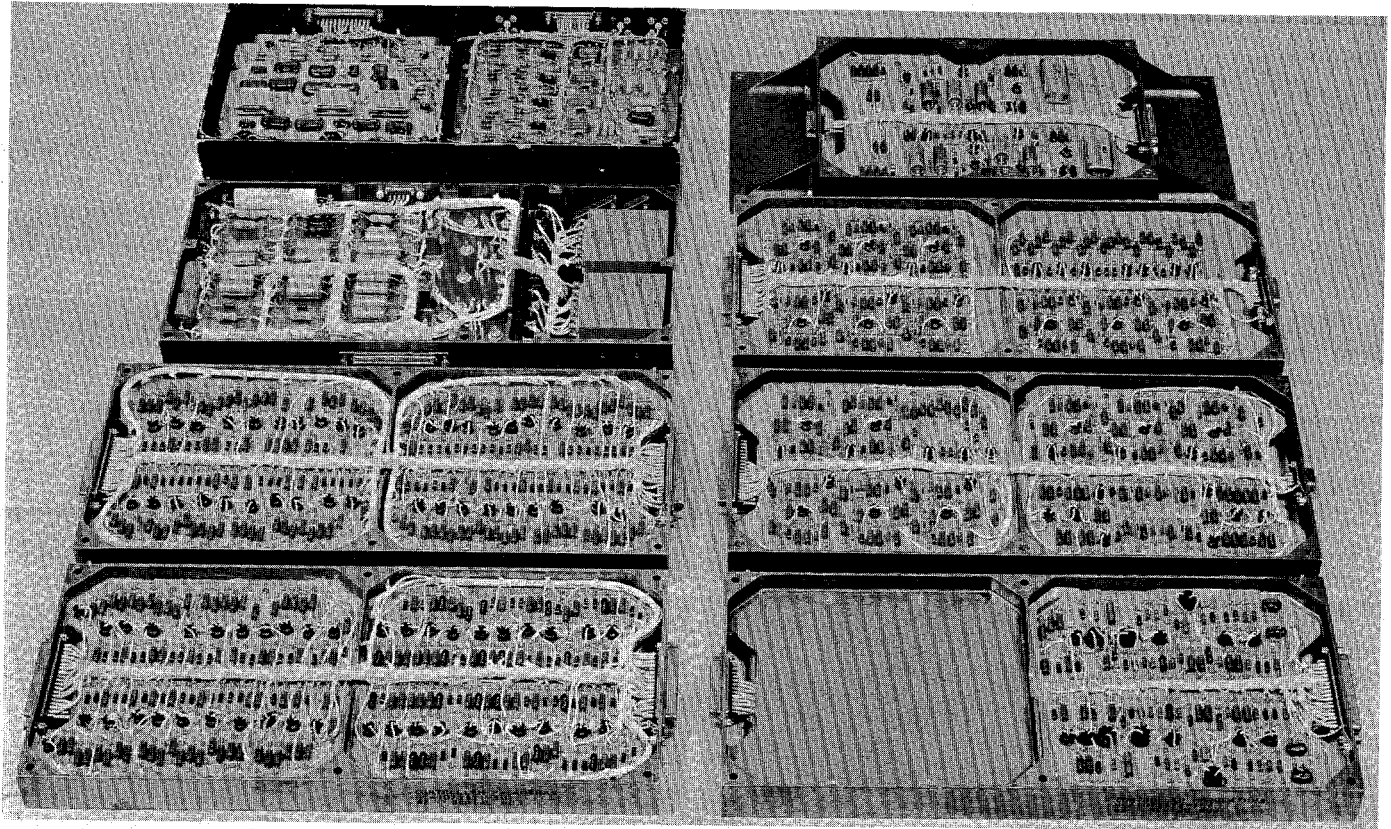


Figure 16. Control programmer and camera sequencer modules



Figure 17. Color wheel

pre-emphasis function was changed to conform with new requirements. Figure 18 shows the pre-emphasis charac-

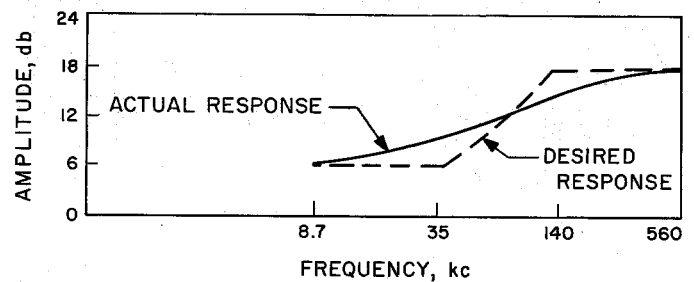


Figure 18. Pre-emphasis function

teristic as introduced by the video combiner. Switching transients at the video combiner output were less than 50 mv in amplitude and less than 3 μ sec wide. The long term drift was measured as $\pm 0.3\%$ between -10 and 65°C .

Stable tone oscillators utilizing quartz crystals have been designed and tested for the three frequencies required for generation of the tone codes. The oscillators show a frequency stability with temperature of 0.09% and a stability amplitude of 11% total over the -10 to 65°C range.

The sequencer and video combiner were tested to verify that the sequencer logic could produce the specified serrated, vertical sync composite video. All three video outputs had a satisfactory composite waveform. Figure 19 shows the pre-emphasis and mixed sync that occur prior to the read period for the Type-P camera. Figure 20 shows the last four lines, or partial Scan 4 read, and the first four lines of the partial-scan group vertical sync (which is a linear mixing of the 144-, 162-, and 180-kc tones).

Random or coherent noise amplitude in the *video-on* periods was less than 10 mv for both partial-scan amplifiers, and less than 25 mv for the full scan summing amplifier. Higher amplitude coherent noise has been attributed to its proximity to the tone oscillator. However, this is not considered a problem. Various combina-

tions of tones of the fully-scanned cameras are shown in Figures 21 through 23.

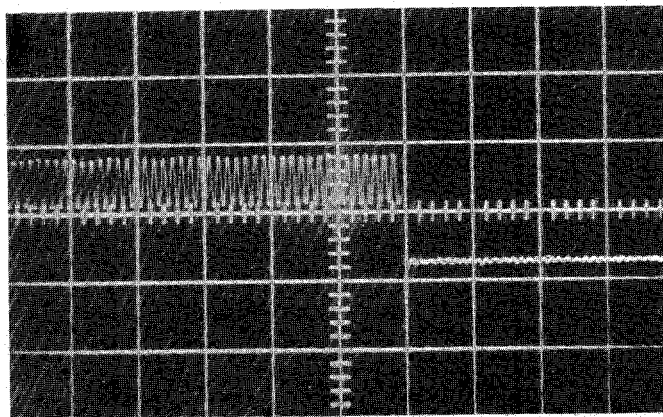


Figure 21. Single tone followed by vertical serration

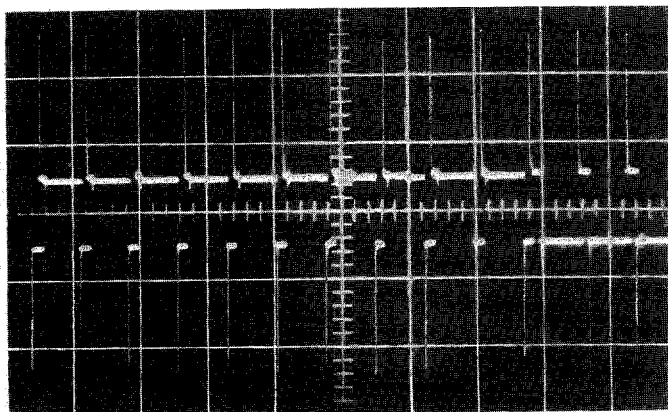


Figure 19. Pre-emphasis of mixed sync on Type-P cameras

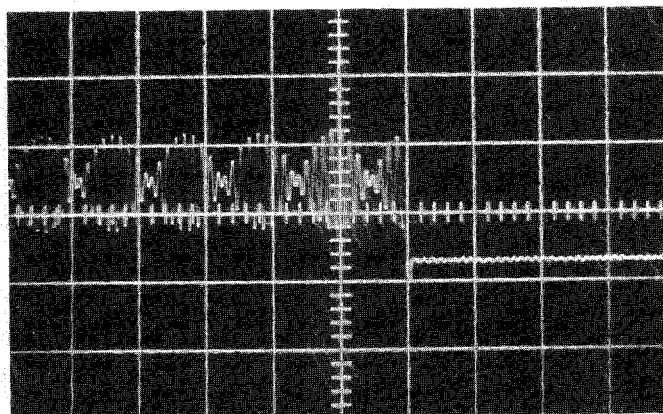


Figure 22. Neutral filter tag

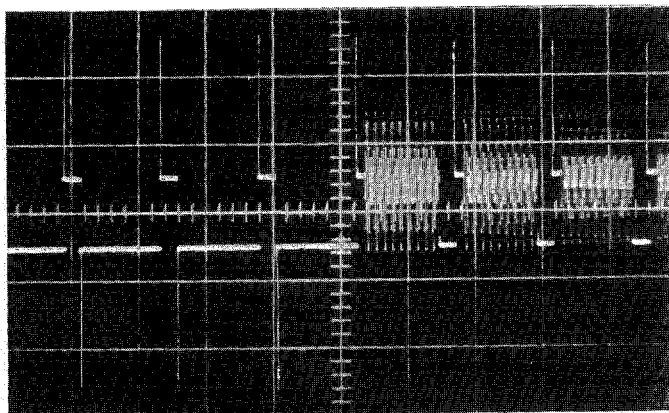


Figure 20. P-4 camera read plus Type P camera vertical sync with tone codes added

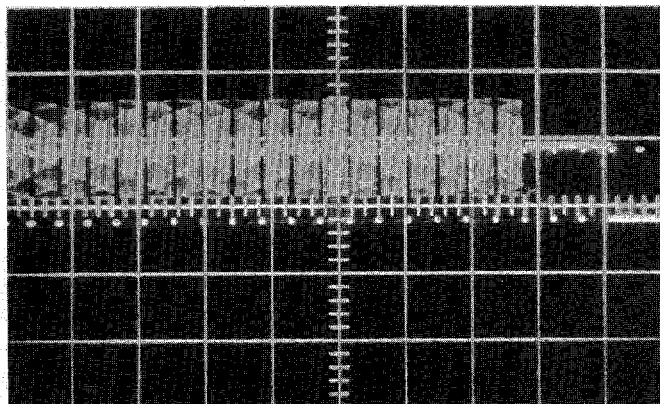


Figure 23. Camera B video serrated vertical sync with tone

D. Telecommunications

During December, final engineering models were assembled and tested. Assembly of qualification test models (QTM) began in mid-December 1961, and one transmitting chain was delivered to AED on January 15, 1962, for testing. The QTM models were subjected to preliminary shock and vibration tests prior to delivery. Preliminary design reviews were completed on all flight equipment during December and January. The measured weight of the QTM was 49 lb as compared with the original estimate of 50 lb.

Measurement of telemetry power consumption indicated that the original estimate was high. The measured current drain was 140 ma during the cruise mode and 350 ma during warmup and terminal operation. Part of the reduction in the estimated power drain can be attributed to the deletion of the three relays.

The telemetry processor is complete and will be used to scale the monitored transmitter-telemetry points to the correct range of 0 to -5 v. Points to be monitored are: the power output, the exciter B+, and the power amplifier temperature, B+ and cathode current. Preliminary vibration tests indicated a need for supporting the transmitter from the top as well as the baseplate. This change, to damp out the mechanical resonance, is being incorporated by AED into the MTM.

The four-port hybrid rings, first received during December, have checked out satisfactorily. The output connector will have to be changed, however, to incorporate a Type-N connector. The first dummy load was received in January. Originally, it had been hoped that this thermal input could be isolated from the rest of the structure, but tests determined that a heat sink will be necessary.

The power amplifier units have continued to operate satisfactorily through all tests. These units have been run up to an 80-w output. The best method of adjusting these units to meet the requirements of the mission is now being studied. The present method requires the power output to be adjusted for a 60-w output after a 5-min warmup period and 5 min of operation. Under this condition the power output starts at about 40 w, increases to 60 w in 5 min, maintains this for the rest of the planned 10-min operation, and then gradually starts to drop off again. After an hour's continuous operation, the power output is down to about 40 w again. This change in power output is attributed to the thermal expansion of the cavities. If the unit is adjusted for maximum power output after the unit reaches thermal equilibrium, then it will

remain constant. This setup method seems to be a good compromise, since the higher battery voltages expected when the unit is first operated tend to compensate for the lower power output. The 60-w output is obtained with the minimum expected battery voltage. At the maximum battery voltage, the transmitter power output is about 80 w after 5 min of operation. Testing of the tube-seal temperatures indicates that a maximum of 130°C is reached after stabilization with a heat-sink temperature of 65°C. An analysis of on-the-launch-pad performance has been made. The results are shown in Table 7. To achieve satisfactory performance, it has been necessary to add a 960-mc preamplifier at the antenna to compensate for cable losses between the antenna and the GSE. The specifications for this equipment require that it have a 30-db gain and a 10-db (max) noise figure. With a 100-mw output from the transmitter, the worst case carrier-to-noise margin above a 12-db threshold is then 2.3 db.

1. System Analysis

a. *Carrier-to-noise ratio.* Calculation of the carrier-to-noise ratio (C/N) is based upon the use of 60-w transmitters, a single four-port hybrid combiner, and a maser receiver (Table 8). At present, there is a 4.2-db margin (worst case) above a 12-db threshold; for *Rangers* RA-5, -6, and -7, the worst case margin is 5.9 db.

Table 7. Results of launch pad performance analysis

Parameter	Value
Transmitter power, dbw	-10 ± 3.0
Spacecraft circuit losses, db	
TV subsystem	-3.3 ± 0.2
Bus	-1.7 ± 0.5
	-0.7
Agena	-5.8 ± 0.5
Space loss to AE building	-111.0 ± 1.0
Polarization loss	-0.5 ± 0.5
Receiving antenna gain	18.5 ± 0.5
Total power received, dbw	-113.8 ± 6.2
	-6.4
System noise temperature, °K (noise figure = 9 ± 1 db)	2020 ± 590
	-480
Noise bandwidth, °K	900 ± 100
Receiver noise power, dbw	-136.0 ± 1.7
	-2.0
Carrier-to-noise ratio (C/N), db	22.2 ± 8.4
	-7.9
Worse case margin above 12-db C/N, db	2.3 —

Table 8. Summary of carrier-to-noise ratio calculations

Parameter	Value
Transmitter power ^a , dbw	18.34 ± 0.5
Spacecraft circuit losses, db	
TV subsystem	3.6 + 0.29
Bus	- 0.14
Bus	1.7 + 0.5
Bus	- 0.7
Spacecraft antenna gain, db	18.1 ± 0.5
Antenna pointing loss, db	0.3 + 0.1
Antenna pointing loss, db	- 0.3
Polarization loss, db	0.14 ± 0.1
Path loss ^b , db	204.2
Receiving antenna gain, db	45.5 ± 0.5
Receiving circuit loss, db	0.4 ± 0.1
Total power received, dbw	-128.4 + 3.24
Total power received, dbw	- 2.59
System noise temperature (including Moon), °K	104 ± 19
Noise bandwidth, mc	1 ± 0.100
Receiver noise power, dbw	-148.41 + 1.2
Receiver noise power, dbw	- 1.4
Carrier-to-noise ratio	20.0 + 4.64
Carrier-to-noise ratio	- 3.79

^aCalculated for 60 w (minimum transmitter power).^bCalculated for 240,000 miles, 960 mc.

b. Picture signal-to-noise ratio. Calculation of signal-to-noise ratio (S/N) is based on a threshold C/N of 12 db, a peak frequency deviation of 200 kc for the video signal, and use of a 185-kc video pass-band filter. The IF bandwidth is 1 mc. A peak-to-peak S/N (rms) of 30 db is used as the criterion of a good picture (Ref 3). The S/N ratio out of the demodulator is:

$$\frac{C}{N} \cdot \frac{\Delta f}{f_m} \left(\frac{3BW_{IF}}{2BW_b} \right)^{\frac{1}{2}} = 11.6 \text{ or } 21.3 \text{ db}$$

where

$$C/N = 3.98$$

$$\Delta f = \text{peak frequency deviation} = 200 \text{ kc}$$

$$f_m = \text{the frequency of the fundamental component} \\ = 185 \text{ kc}$$

$$BW_{IF} = \text{IF bandwidth} = 1000 \text{ kc}$$

$$BW_b = \text{bandwidth of the pass-band filter} = 185 \text{ kc}$$

Converting from an rms ratio to a peak-to-peak signal (S_{pp}) to rms noise (N_{rms}) ratio gives 30.3 db. With a worst case C/N margin of 4.2 db, the picture quality is 34.5 db. This exceeds the 30-db picture criterion by 4.5 db.

An alternate approach is to use pre-emphasis to provide the required picture S_{pp}/N_{rms} level with a reduced IF bandwidth. Pre-emphasis attenuates the line frequency by 12 db but this is followed by a flat 6-db gain, so that the effective peak deviation is 100 kc. The required IF bandwidth is reduced by 200 kc. This reduced bandwidth provides more adjacent channel rejection, and thus reduces the effects of cross-talk between channels. The use of pre-emphasis in this manner has been found to give an effective improvement of 6.6 db, as reported in the original proposal. At threshold ($C/N = 12 \text{ db}$) the

$$\frac{S_{pp}}{N_{rms}} = 29.1 \text{ db}$$

However, the threshold of this system is 1.1 db lower than the 1-mc bandwidth system. Therefore, for the same received signal power as the 1-mc bandwidth system, at threshold, the S_{pp}/N_{rms} is 30.2 db. The net result is that the use of pre-emphasis provides a lower threshold, and less cross-talk between channels.

c. Terminal telemetry. Since the telemetry is a PAM/FM/FM system there are two FM threshold points to consider. The first is the C/N ratio into the first demodulator. The subcarrier S/N (S_{sc}/N_{sc}) out of the first demodulator must exceed the 12-db threshold of the second demodulator. The S_{sc}/N_{sc} ratio is given as (Ref 4)

$$\frac{C}{N} \cdot \frac{\Delta f}{f_{sc}} \left(\frac{BW_{IF}}{2BW_{sc}} \right)^{\frac{1}{2}}$$

Using

$$\frac{C}{N} = 12 \text{ db}$$

$$f_{sc} = 225 \text{ kc}$$

$$\Delta f = 35 \text{ kc}$$

$$BW_{IF} = 1 \text{ mc}$$

$$BW_{sc} = 10 \text{ kc}$$

then

$$\frac{S_{sc}}{N_{sc}} = 4.15 \text{ or } 12.3 \text{ db}$$

The PAM signal out of the second demodulator has a S/N ratio as determined by:

$$\frac{S_{sc}}{N_{sc}} \cdot \frac{\Delta f}{f_m} \left(\frac{3BW_{sc}}{2f_m} \right)^{\frac{1}{2}} = 2.96 \times 10^4 \text{ or } 89.4 \text{ db}$$

using

$$\Delta f = 5 \text{ kc}$$

$$f_m = 20 \text{ cps}$$

$$BW_{sc} = 10 \text{ kc}$$

For a 1% data resolution, the required peak signal-to-peak noise is 40 db. Using a crest factor of 4 for the noise (corresponding to a probability of 99.99% that the noise peaks will not exceed 4 times the rms noise voltage), the S/N ratio is $40 + 12 - 3$ or 49 db. Thus, the terminal telemetry channel has a 40-db performance margin when the C/N ratio is at the 12-db threshold. However, due to high deviation of the subcarrier, the required threshold signal is actually greater than 12 db. Nevertheless, it is felt that the actual S/N will exceed 50 db.

During the emergency telemetry mode of operation, the receiver bandwidth is switched to 100 cps. The switch from a 1-mc IF to a 100-cps IF reduces the noise power by 40 db. Thus, the received signal power can drop by 48 db from the normal signal level and still exceed a 12-db threshold. This corresponds to using an antenna that is 29.9 db below isotropic.

For a 12-db threshold signal, the telemetry data has a S/N ratio of:

$$\frac{C}{N} \cdot \frac{\Delta f}{f_m} \left(\frac{3BW_{IF}}{2f_m} \right)^{\frac{1}{2}} = 460 \text{ or } 53.3 \text{ db}$$

using

$$\begin{aligned} \Delta f &= 50 \text{ cps} \\ BW_{IF} &= 100 \text{ cps} \\ f_m &= 3 \text{ cps (fundamental component)} \end{aligned}$$

This S/N would give emergency data that has better than 1% resolution.

2. Channel 8 Telemetry

For cruise-mode telemetry, the same performance requirement as used for terminal telemetry is assumed. That is, the desired output S/N is sufficient to give 1% accuracy. This means that the peak signal-to-peak noise ratio must be at least 40 db (using a crest factor of 4 for the noise); i.e., the noise peaks will be less than 4 times the rms value of the noise 99.99% of the time. The rms signal-to-rms noise ratio must be $40 + 12 - 3$ or 49 db. The subcarrier-to-noise ratio in a 100-cps band centered at 3 kc is assumed to be at least 8 db (a conservative estimate for a threshold signal level with a phase lock discriminator). Since the base-band information in channel 8 is 1.5 cps and the peak deviation is 40 cps, the resultant signal-to-noise (S/N) out of the discriminator is

$$= \sqrt{3} \frac{\Delta f}{f_m} \left(\frac{BW_{sc}}{2f_m} \right)^{\frac{1}{2}} \frac{C}{N} = 56.6 \text{ db}$$

using

$$\begin{aligned} \Delta f &= 40 \text{ cps} \\ f_m &= 1.5 \text{ cps} \\ BW_{sc} &= 100 \text{ cps centered at 3 kc} \\ \frac{C}{N} &= 8 \text{ db or } 2.5 \end{aligned}$$

Thus the required performance of 49 db is exceeded by 7.6 db when the threshold subcarrier-to-noise ratio of 8 db is received from the bus.

A composite telemetry is provided to the TV subsystem by the bus and a subcarrier band-pass filter is located in the subsystem equipment to select the Channel 8 telemetry.

a. Signal-to-noise ratio for command tones. The following values were used in calculating signal-to-noise ratio for the tone codes:

- (1) C/N at threshold: 10 db or 3.17
- (2) Tone frequencies: 144 kc, 162 kc, 182 kc
- (3) Tone filter bandwidth: 500 cps
- (4) IF bandwidth: 900 kc
- (5) Deviation due to tones: ± 20 kc (Normal deviation for TV signal is ± 200 kc for ± 0.5 v. Since the tones are ± 0.05 v, the deviation due to the tones is ± 20 kc, i.e., $\Delta f_t = 20$ kc)

For an FM system with subcarriers that have a base band small in comparison with the total base band, the S/N is given by:

$$\frac{C}{N} \cdot \frac{\Delta f_t}{f_{sc}} \left(\frac{BW_{IF}}{2BW_T} \right)^{\frac{1}{2}}$$

Therefore, the S/N for the 144-kc tone is 13.2 or 22.4 db; for the 162-kc tone, the S/N ratio is: 11.7 or 21.3 db; and for the 188-kc tone, the S/N ratio is: 10.1 or 20 db.

Through the use of pre-emphasis, the effective deviation of the tones is increased by at least 6 db. This results in a S/N for the tones of 26 to 28.4 db when the C/N is at the 10-db threshold.

b. The effect of pre-emphasis. This analysis is based on the use of a minimum phase shift network with a 12-db separation between the plateaus for the pre-emphasis and de-emphasis circuits. For ease of analysis the frequency will be scaled such that the following transfer functions may be used (Fig 24). The frequency and time scale factors will be discussed after examining the effects of pre-emphasis on some pulses. The system used for analysis is shown in Figure 25.

The sum of Gain A and Gain B is 12 db, which compensates for the losses in the emphasis networks and thus give a flat system response; i.e., the transfer function of the system for the signal is unity. First, the effects of the synchronizing pulse width and rise time are studied in order to determine how the gain may be split between A and B. Also, the effect on partially-scanned video is examined. Then the effect on S/N is determined.

c. Effect of synchronizing pulse width and rise time on deviation. The deviation limit of 1 v peak-to-peak is to be maintained, and thus the size of any spikes produced by pre-emphasis must be determined. First, the pulse shown in Figure 26a will be analyzed. For analysis this pulse will be represented by the sum of two step functions as shown in Figure 26b. The Laplace transform of this is:

$$F(S) = \frac{1}{S} (1 - e^{-110S})$$

The output of the pre-emphasis network is:

$$\begin{aligned} E_o(S) &= F(S) \cdot G(S) = \frac{1}{S} \left[\frac{(S+1)}{(S+4)} \right] (1 - e^{-110S}) \\ &= \frac{1}{S} \cdot \frac{(S+1)}{(S+4)} - \frac{1}{S} \cdot \frac{(S+1)}{(S+4)} e^{-110S} \\ &= A(S) - A(S) e^{-110S} \\ e_o(t) &= a(t) - a(t - 110) \end{aligned}$$

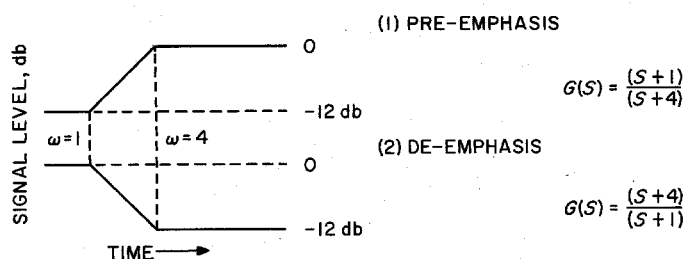


Figure 24. Frequency break points

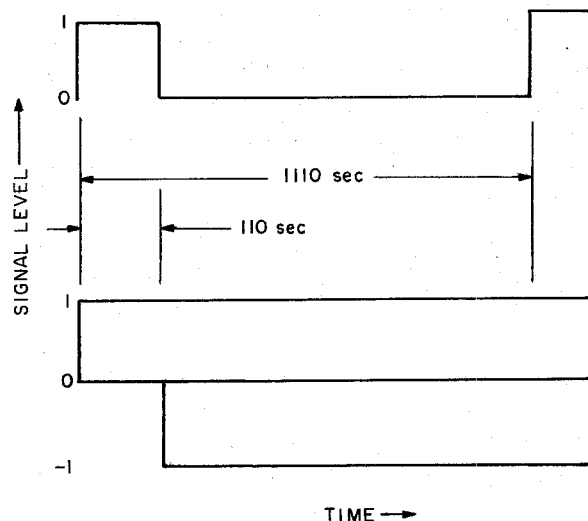


Figure 26. Synchronizing pulse

$$\begin{aligned} A(S) &= \left[\frac{1}{S} \cdot \frac{(S+1)}{(S+4)} \right] \\ &= \frac{1}{4S} + \frac{3}{4(S+4)} \\ a(t) &= \frac{1}{4} \mu(t) + \frac{3}{4} e^{-4t} \\ e_o(t) &= \frac{1}{4} \mu(t) + \frac{3}{4} e^{-4t} - \frac{1}{4} \mu(t - 110) - \frac{3}{4} e^{-4(t-110)} \end{aligned}$$

This equation is plotted in Figures 27 and 28.

If the pre-emphasis circuit is followed by a flat amplifier with a 6-db gain and then a limiter which limits the signal to 1 v peak-to-peak it appears as shown in Figure 29. For analysis, this pulse will be represented by a 0.75-v step at $t = 0$; a ramp at $t = 0.4$ with a slope of $-\frac{1}{2}$ v/sec; and a second ramp at $t = 0.9$ with a slope of $\frac{1}{2}$ v/sec. This representation of the pulse is shown in Figure 30. The Laplace transform of this voltage is given by:

$$E_1(S) = \frac{3}{4} \cdot \frac{1}{S} - \frac{1}{2} \cdot \frac{1}{S^2} \cdot e^{-0.4S} + \frac{1}{2} \cdot \frac{1}{S^2} \cdot e^{-1.4S}$$

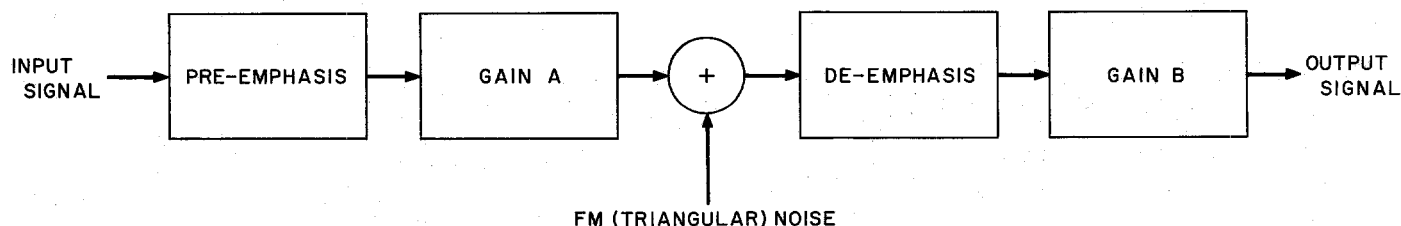


Figure 25. System used for analyzing effects of emphasis

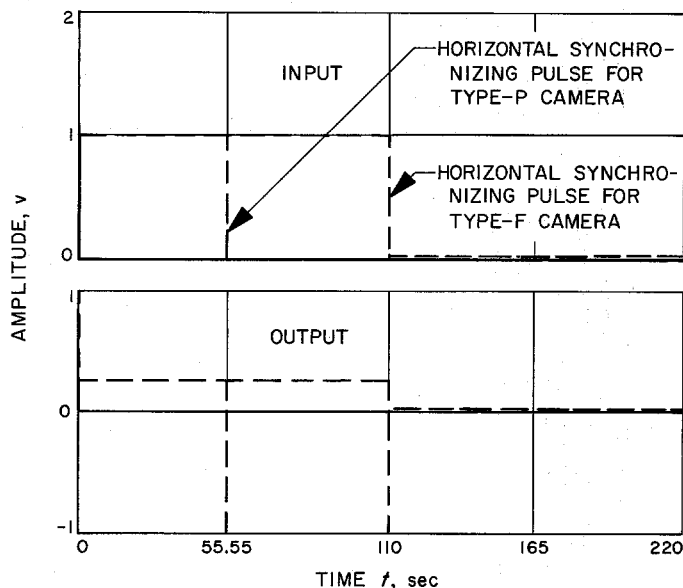


Figure 27. Effect of pre-emphasis on synchronizing pulses with zero rise time

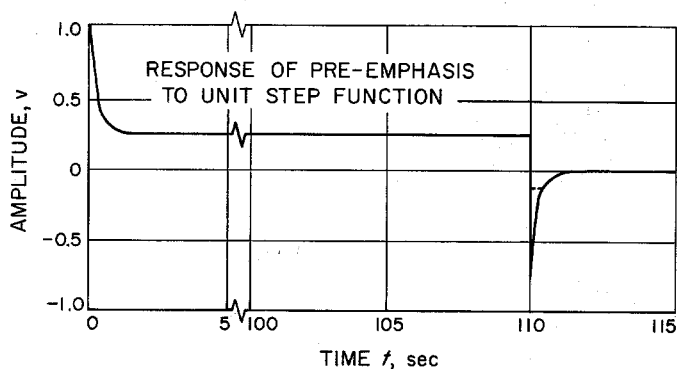


Figure 28. Effect of pre-emphasis on zero rise time synchronizing pulse (expanded time scale)

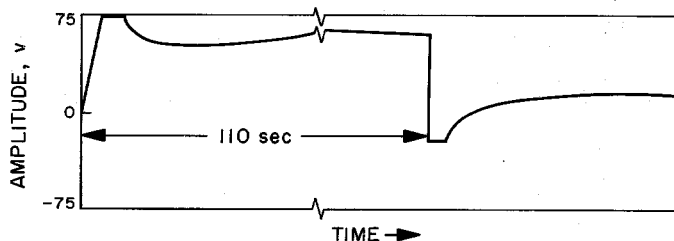


Figure 29. Synchronizing pulse after pre-emphasis, amplification, and limiting

If this voltage is now passed through the de-emphasis network, the output voltage is:

$$E_2(S) = \left(\frac{3}{4} \cdot \frac{1}{S} - \frac{1}{2} \cdot \frac{1}{S^2} \cdot e^{-0.4S} + \frac{1}{2} \cdot \frac{1}{S^2} \cdot e^{-0.9S} \right) \frac{(S+4)}{(S+1)}$$

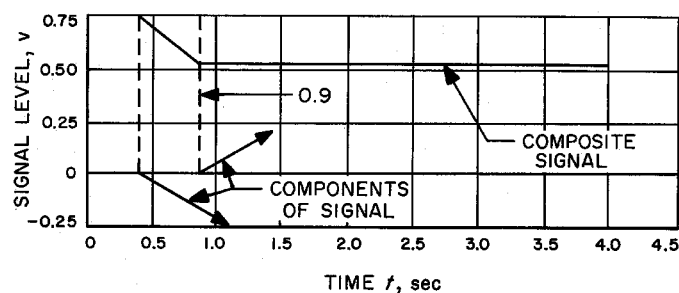


Figure 30. Graphic analysis of pre-emphasized, amplified, and limited synchronizing pulse

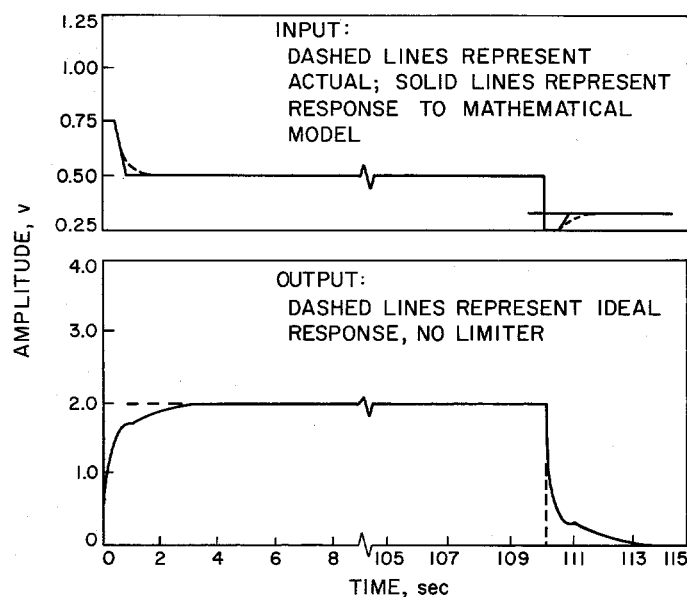


Figure 31. Effect of pre-emphasis followed by gain, limiting, and de-emphasis

$$\begin{aligned} e_2(t) &= \frac{3}{4} 4\mu(t) - 3e^{-t} \\ &\quad - \frac{1}{2} [4(t-0.4) - 3\mu(t-0.4) + 3e^{-(t-0.4)}] \\ &\quad + \frac{1}{2} [4(t-0.9) - 3\mu(t-0.9) + 3e^{-(t-0.9)}] \end{aligned}$$

This result is plotted in Figure 31. This shows that the rise time (10 to 90%) is limited to 1.4 sec. Next, the effect of limiting rise time on the output of pre-emphasis will be analyzed. The ramp will be limited to (a) a rise time of 1 sec, and (b) a 2-sec rise time. Scaling these rise times for analysis results in a ramp of 1 v/sec and 0.5 v/sec, respectively. The two pulses are separated into components for analysis as shown in Figure 32. The inputs of Figure 32 are represented mathematically by:

$$E_1(S) = \left[\frac{1}{S^2} - \frac{1}{S^2} e^{-S} \right]; \text{ Fig 32a}$$

$$= 0.5 \left[\frac{1}{S^2} - \frac{1}{S^2} e^{-2S} \right]; \text{ Fig 32b}$$

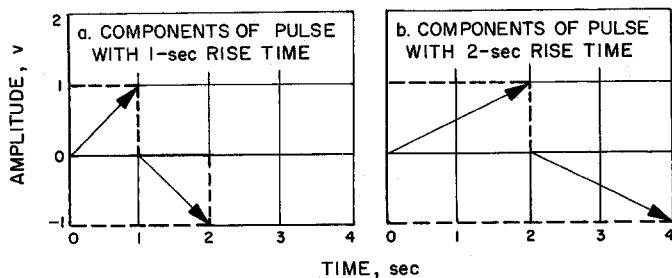


Figure 32. Analysis of pulses to determine effects of limiting rise time

The output of the pre-emphasis circuit for an input of A, Figure 32, is:

$$\begin{aligned}
 E_o(S) &= \left[\frac{1}{S^2} - \frac{1}{S^2} e^{-s} \right] \frac{(S+1)}{(S+4)} \\
 &= (1 - e^{-s}) \left[\frac{1}{4} \cdot \frac{1}{S^2} + \frac{3}{16} \cdot \frac{1}{S} - \frac{3}{16} \frac{1}{(S+4)} \right] \\
 e_o(t) &= \left[\frac{1}{4} t + \frac{3}{16} \mu(t) - \frac{3}{16} e^{-4t} \right] \\
 &\quad - \left[\frac{1}{4} (t-1) + \frac{3}{16} \mu(t-1) - \frac{3}{16} e^{-4(t-1)} \right]
 \end{aligned}$$

The output of the pre-emphasis circuit for an input of B, Figure 32, is:

$$\begin{aligned}
 E_o(S) &= \frac{1}{2} \cdot \frac{1}{S^2} [1 - e^{-2s}] \frac{(S+1)}{(S+4)} \\
 e_o(t) &= \frac{1}{2} \left[\frac{1}{4} t + \frac{3}{16} \mu(t) - \frac{3}{16} e^{-4t} \right] \\
 &\quad - \frac{1}{2} \left[\frac{1}{4} (t-2) + \frac{3}{16} \mu(t-2) - \frac{3}{16} e^{-4(t-2)} \right]
 \end{aligned}$$

These results are plotted in Figure 33. As can be seen from Figure 33, a 6-db flat gain following pre-emphasis of a synchronizing pulse, rising from the full white level with a rise time of 2 sec, would not exceed the peak deviation limits. Decreasing the rise time to 1 sec would cause the transient spikes to exceed the 1-v peak-to-peak deviation level by 40%.

From this analysis it can be seen that the transient conditions at the leading and trailing edges of the synchronizing pulse are short compared to the pulse width. Transient conditions die out at a rate determined by the denominator of the pre-emphasis transfer function. Allowing five time constants for the transient, and adding the rise time of the input pulse, gives the minimum half-amplitude pulse width that may be passed and still have the first statement of this paragraph hold true. When this statement is true, the effects of the leading and trailing edges are independent of each other.

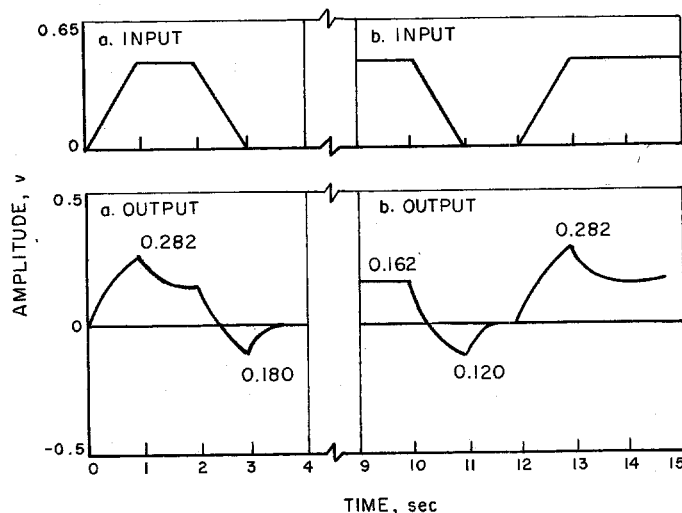


Figure 33. Effect of pre-emphasis on single video pulse

For the *Ranger* TV subsystem, the rise time of the synchronizing pulse will be approximately 2 μ sec. This is based on a 200-kc pass band which will pass a rise time of $t_r = 0.35/f(mc) = 0.35/0.2 = 1.75 \mu$ sec (Ref 5).

The pulse width of the horizontal synchronizing pulse for the Type-P camera is 111.1 μ sec; for the Type-F camera, the synchronizing pulse is 220 μ sec wide. If a time scale is chosen in which 1 sec represents 1 μ sec, then, with a synchronizing pulse having a 2- μ sec rise time as the input to a pre-emphasis network, with the transfer function

$$G_p(S) = \frac{(S + 10^6)}{(S + 4) 10^6}$$

the output of the pre-emphasis network would be as shown in Figure 33, B. This would put the frequency break points at 160 and 640 kc. Experimentally, it has been observed that for the *Ranger* TV subsystem the break frequencies could lie at 35 and 140 kc without having transients that exceed the level shown in Figure 33, B. This observation may be explained by the fact that the leading edge is not actually a ramp and does not have the high frequency components of a ramp. Experimentally, the total transition period lasted 8 μ sec for a step function with a 2- μ sec rise time. Thus, the same pre-emphasis network could be used for Type-P and Type-F synchronizing pulses. With the 3-db points located at 35 and 140 kc, the gain after pre-emphasis could be increased 8 db without exceeding a 1-v peak-to-peak deviation level.

d. *Effect of video components on deviation.* For video components that can be considered as pulses with rise times limited to 2 μ sec, the preceding analysis applies,

provided the transients have time to die out. Using the experimental data that the total transition period takes $8 \mu\text{sec}$, then the transients in a pulse with a width less than $8 \mu\text{sec}$ would die out in one period. Thus, repetitive signals up to 64 kc would not take more than one cycle to reach a steady-state dc level. Since the video has an amplitude of 0.65 v peak-to-peak after pre-emphasis, the levels would be -0.120 to $+0.282$ v, as shown in Figure 34.

After pre-emphasis, a higher frequency signal, such as the limiting resolution 200-kc sine wave shown in A of Figure 35, would require several periods to reach the steady-state condition shown in B of Figure 35. However, since the camera output at 200 kc is only 40% of the low-frequency response, the peaks of a steady 200-kc signal would be at -0.95 and 0.260 v in the worst case. Thus, it is felt that, because of the random nature of video signals, and the high frequency roll-off in the camera, the worst case for peak signals is represented by the synchronizing pulse to white level.

e. Summary of the effects of pre-emphasis on deviation. The aim of the preceding analysis was to determine how much gain can be used after pre-emphasis without exceeding a 1-v peak-to-peak deviation level. Before pre-emphasis, the signal has a maximum excursion from zero (white) to 1 v (tip of synchronizing pulse). After pre-emphasis, because of the high frequency video roll-off in the same camera, the peak excursions are still limited by the maximum transitions of the synchronizing pulse but the limits in the worst case are now -0.12 and 0.34 v. Video excursions range from -0.120 to 0.280 v. Since the

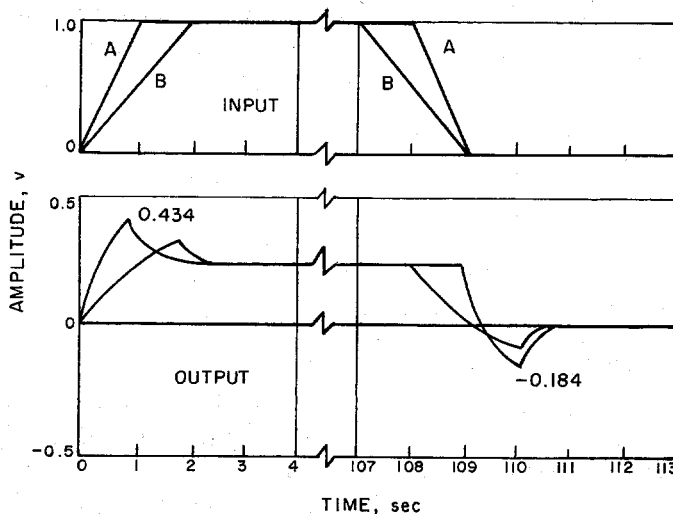


Figure 34. Effect of pre-emphasis with limited rise time on synchronizing pulses

actual signals will not possess the high frequency components of the ramp functions used in the analysis, it is felt that the pre-emphasis can be followed by at least 8-db gain before going into the modulator.

f. Signal-to-noise improvement. In this section, the effect of an 8-db boost in high frequency signals and a 4-db loss in low frequency signals will be discussed. In order to regain the transmitted signal, highs will have to be attenuated by 12 db, but the over-all signal will have to be amplified by 4 db in order to regain the original signal.

Figure 36 shows the change in the noise spectrum. The total noise power in the original spectrum is given by:

$$N_p + K \int_{-200}^{200} \left(\frac{f}{200} \right)^2 df = K \left[\frac{f^3}{3(200)^2} \right]_{-200}^{200} = 2K \frac{200}{3}$$

The total noise power after de-emphasis is given by:

$$\begin{aligned} N_p &= 2K \left(\int_0^{35} \left(\frac{1.58f}{200} \right)^2 df + 35 \int_{35}^{140} (0.28)^2 df \right. \\ &\quad \left. + \int_{140}^{200} \left[\frac{1}{2.52} \left(\frac{f}{200} \right) \right]^2 df \right) \\ &= \frac{2K}{3} (50.4) \end{aligned}$$

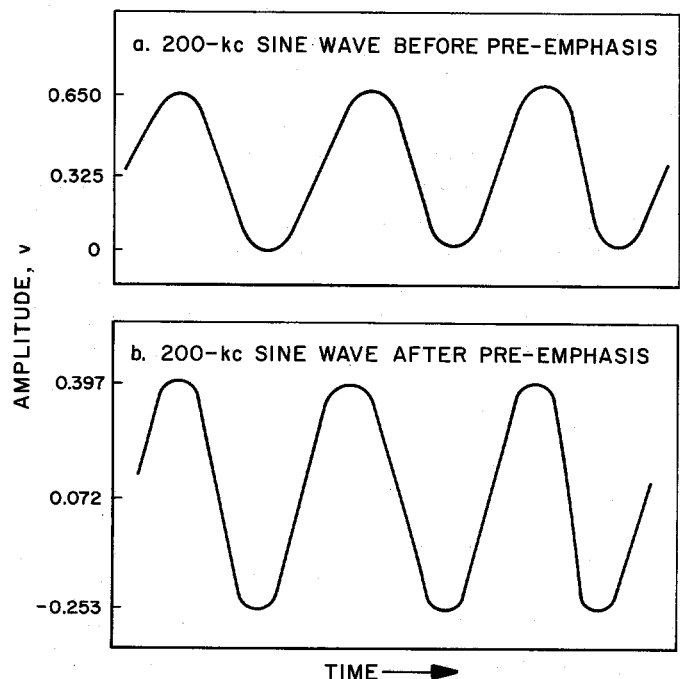


Figure 35. Effect of pre-emphasis on a 200-kc sine wave

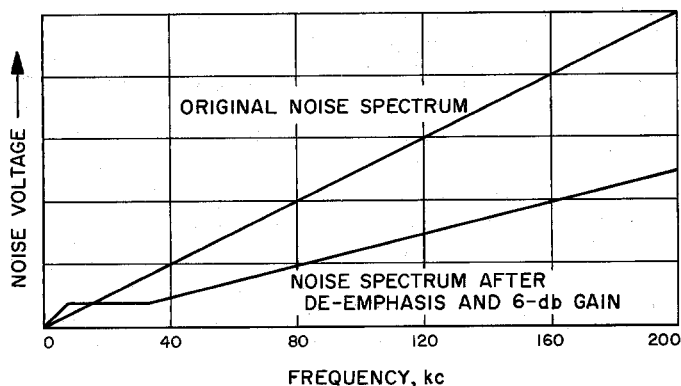


Figure 36. Noise spectrum

The total noise power is reduced by:

$$\frac{2K(200)^3}{2K(50.4)^3} = 3.98 \text{ or } 6 \text{ db}$$

g. Effect on receiver bandwidth and FM improvement. Because the peak deviation for the fundamental components of the horizontal line frequency is reduced by 4 db, the required IF bandwidth is reduced. When 8 db of gain is used following pre-emphasis, the IF pass band need only be $2[(\Delta f/1.59) + fm]$ where Δf is the peak deviation for a nonpre-emphasized video signal. For the TV subsystem program, this means that the 900-kc bandwidth requirement may be reduced by 150 kc. This would reduce the required threshold signal power by about 1 db.

A sacrifice is made when pre-emphasis is used. It has been shown that pre-emphasis does give an appreciable S/N improvement and reduces the required IF bandwidth. However, the price paid is in the reduction of the FM improvement. For a 6-db improvement by pre-emphasis, the FM improvement is reduced by 4.9 db. This reduction in FM improvement may be calculated by:

$$\text{Reduction in FM improvement} = D_1 \left[\left(\frac{M+1}{D_1} \right) \right]^{1/2}$$

where

$$D_1 = \frac{\text{deviation for nonpre-emphasized video}}{\text{deviation for pre-emphasized video}}$$

M = modulation index for nonpre-emphasized video

h. Conclusions. Pre-emphasis can be used to gain a S/N improvement for video transmission over an FM system. The amount of improvement possible is limited by the shape of the video spectrum from the cameras,

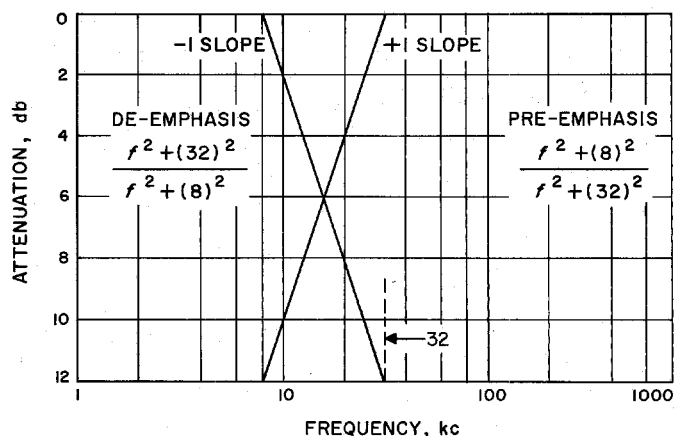


Figure 37. Straight line approximations of pre-emphasis and de-emphasis characteristics

the rise times and shapes of the horizontal synchronizing and blanking pulses, and the system bandwidth. FM improvement can be sacrificed for pre-emphasis improvement to reduce transmission bandwidth and required transmitter power.

On the *Ranger* TV subsystem program, the same pre-emphasis network can be used for the fully- and partially-scanned cameras. The transfer function of the recommended network is shown in Figure 37. The amount of flat gain to be used after pre-emphasis should be 8 db. The rise time of synchronizing pulses should be limited to 2 μ sec. The video response of the *Ranger* cameras at 200 kc should be down to 40% of their low frequency response. The de-emphasis transfer function is also shown in Figure 37. The flat gain associated with de-emphasis is 4 db in order to return to the 1-v peak-to-peak level of the original signal. Amplitude limiting following pre-emphasis is to be avoided, since it would make rise time dependent upon signal level.

E. Power Supplies

1. Batteries

Qualification batteries Serial Nos. 5 and 6 were received. Serial No. 6 successfully completed a simulated mission at low temperature, and is now undergoing qualification testing. Serial No. 5 (life test model) was found to have low resistance to ground, and was returned to ESB. The failure was due to improper sealing, and steps were taken to avoid future failures. Battery design has

been frozen (including batteries Serials Nos. 5 to 8). Maximum allowable battery weight is 43.5 lb. Figure 38 illustrates the engineering model battery.

2. Voltage Regulator

Using silicon-controlled rectifiers for activating the load at warmup necessitated a separate regulator for supplying cruise mode power. The regulator was designed and a breadboard was built. Design of both regulators was tentatively released and engineering models were built. After tests and minor mechanical and electrical changes, the design was frozen. Prototypes will be completed shortly.

Figure 39 illustrates the engineering model voltage regulator. The base of the cruise-mode regulator was redesigned to strengthen the package, and reduce the possibility of setting up adverse vibrational modes. Both low- and high-current engineering model regulators were constructed and tested. Table 9 shows low-current

Table 9. Low current regulator performance

Temperature, °C	Voltage in, v	Load current, amp		
		0.1304	0.250	0.342
		Load voltage, v		
25	30	28.136	27.786	27.479
	33	28.163	27.821	27.509
	36	28.190	27.849	27.543
	39	28.213	27.875	27.570
	42	28.285	27.898	27.600
65	30	27.996	27.641	27.358
	33	28.027	27.678	27.399
	36	28.054	27.712	27.432
	39	28.077	27.739	27.465
	42	28.103	27.770	27.502
0	30	28.258	27.899	27.594
	33	28.294	27.930	27.626
	36	28.319	27.968	27.658
	39	28.338	27.988	27.692
	42	28.364	28.020	27.710

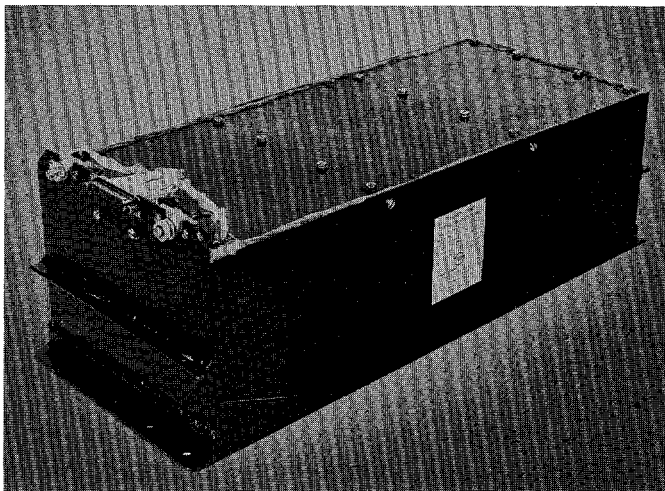


Figure 38. Engineering model battery

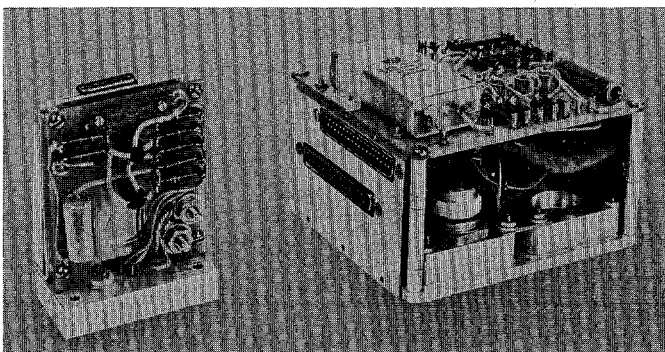


Figure 39. Engineering model voltage regulator

Table 10. Open loop gain data for low current regulator

Frequency kc	Phase shift, deg	Gain		Voltage, v	
		db	V_{out}/V_{in}	in	out
0.010	180	57	700	0.001	0.7
0.050	180	57	700	0.001	0.7
0.100	180	54	500	0.001	0.5
0.500	160	56	620	0.001	0.62
1	150	50	320	0.001	0.32
3	130	44	160	0.001	0.16
6	125	33.2	46	0.003	0.14
10	125	27	23	0.003	0.07
30	120	8	2.5	0.008	0.02
60	95	2	1.3	0.012	0.016
88	90	-13	0.22	0.013	0.003
100	80	-11	0.28	0.014	0.04
130	65	-17	0.14	0.014	0.02

regulator performance, depicting regulation versus input voltage and load current at selected temperatures. Table 10 lists open-loop gain data of the low-current regulators.

Table 11 shows high-current regulator performance, depicting regulation versus input voltage and load current at selected temperatures. Table 12 lists closed-loop gain data of the high-current regulator. Transient response was measured by increasing current by 1 amp. This caused a change in the bias voltage of 0.09 v and the transient recovered to 0.07 v within 2 msec.

Table 11. High current regulator performance

Temperature, °C	Voltage in, v	Load current, amp		
		0.5	2.5	5.0
		Load voltage, v		
25	29.5	27.509	27.507	27.450
	33	27.514	27.511	27.505
	36	27.514	27.611	27.506
65	29.5	27.584	27.577	27.353
	33	27.582	27.581	27.573
	36	27.585	27.581	27.574
0	29.5	27.485	27.486	27.448
	33	27.491	27.489	27.486
	36	27.492	27.489	27.487

Table 12. High current regulator, open loop gain

Frequency, kc	Loop gain	Voltage, v	
		in	out
0.010	1.0	0.05	0.05
0.050	1.0	0.05	0.05
0.100	1.0	0.05	0.05
0.500	0.6	0.05	0.03
1	0.6	0.05	0.03
3	0.6	0.05	0.03
6	0.6	0.05	0.03
10	0.6	0.05	0.03
30	0.6	0.05	0.03
60	0.6	0.05	0.03
80	0.7	0.05	0.035
82	1.0	0.065	0.065
85	1.07	0.07	0.075
90	0.9	0.05	0.045
100	0.8	0.05	0.04
150	0.6	0.05	0.03
200	0.6	0.05	0.03
250	0.6	0.05	0.03
300	0.2	0.05	0.01
400	0.2	0.05	0.01

NOTE: Regulator tested with 5-amp load and 32.0 v battery voltage. For a stable system the peak of the closed loop gain should be below 1.25.

Prototype qualification and life test models are being constructed and will be tested when completed. Terminal-mode regulator weight has been reduced by using magnesium in most of the structural members, and using beryllium oxide insulators (reducing size of the heat sinks). The structure has been arranged so that the major heat producing units are mounted almost directly to the basic structure. (Two regulator base surfaces are utilized for this purpose.) Cover design permits examination or

removal of most of the regulator components without removal from the vehicle.

3. Ground Support Equipment

The design philosophy for the GSE launching pad area equipment has been changed to enable locating the battery charger in the launching shed, and to locate the chart recorder, monitor and control panel in the spacecraft bus rack in the blockhouse.

F. Thermal Control

The detailed thermal analysis, conducted to determine the continuous temperature-time history of the electronic modules throughout the mission, continued during this period. As a result of this study, a major design change added four circular-ring fins to the existing shield at the bulkhead attaching points. Design and construction of the thermal test model, containing simulated electronic modules, is nearly complete.

A study is being conducted that will enable any required solar absorptivity from 0.2 to 0.9 to be obtained by simply specifying the paint mixture ratio, and a study has been completed to determine the influence of radiation coupling versus conduction coupling. It was concluded that radiation coupling factors are of relatively minor importance.

1. Design Concept

The thermal design is based upon obtaining a uniform temperature on the thermal shield during cruise. All interior components see this temperature field and arrive at this value at steady-state with no internal power. The shield temperature equation is as follows:

$$\sigma T^4 = S \left(\frac{a}{A} \right) \left(\frac{\alpha}{\epsilon} \right)$$

where

S = solar constant

a = projected area of shield to solar field

A = surface area of shield

α = solar absorptivity of shield

ϵ = thermal emissivity of shield

σ = Stefan-Boltzmann radiation constant

T = temperature

The only variable is the α/ϵ ratio, since the geometry of the subsystem is firmly established. (Due to the extreme geometry of the shield, the ratio a/A is very small.) Thus, to maintain a shield temperature of 10°C during cruise, a ratio of α/ϵ of approximately 2.5 is required. Polished metals are the primary materials that have this ratio. However, a design study showed that polished metals have peculiar properties such as follows:

- (1) Ratios of hemispherical-to-normal emissivity approaching $4/3$.
- (2) Reflectivity strongly increasing with increased angle of incidence as measured from the normal to the surface.
- (3) Absolute values of α and ϵ dependent upon the degree of polish, oxidation, type of metal, fingerprints and scratches, etc.

It was concluded that the design value of α/ϵ is not easy to obtain, and cannot be predicted to the degree required for reliable thermal control. The most unknown parameter was the reflectivity of solar energy at high angles of incidence (83.75°) for the shield surface during cruise. To allow the design to continue with such an unknown was considered to be extremely risky.

2. Circular Ring Fins

Thus, major design changes have added four circular-ring fins to the existing shield at the bulkhead attachment points. The fin surfaces are normal to the solar field and are dimensioned so that the surface area of the shield is completely shadowed during cruise (eliminating the angular-incidence problem). The surface finishes are paints whose spectral properties can be easily measured and are not adversely affected by oxidation or environment. A thermal analysis indicated that shield temperature can be maintained at the desired cruise temperature with maximum temperature gradients of less than 4°C on the shield. Addition of fins revised the proposed method of test simulation for the thermal test model. The previous test plan proposed that infrared lamps be used for approximating solar heating. This method was deemed to be impractical for the model with fins, since the infrared sources would require precise collimation. Simulation of solar heating will now be accomplished with electrical-contact heating rings bonded to the solar illuminated areas (Fig 40). This type of simulation will enable precise determination of the solar absorptivities

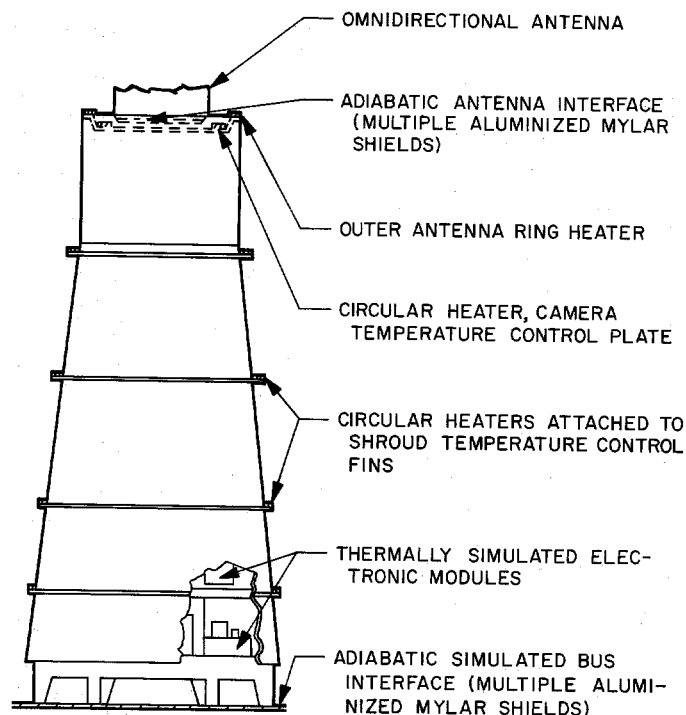


Figure 40. Location of electrical heaters simulating absorbed solar energy

required to produce the desired thermal operation of the subsystem.

To improve the thermal contact of the fins to the shield, it was proposed that the fins be attached with an epoxy bond in addition to the machine screw supports. A test was conducted to determine bonding material conductance. Eccobond 45 epoxy cement was chosen because of its elastic property which resists cracking when the shield is flexed with the fins in place. The test indicated a conductance of approximately $2.5 \text{ w/in.}^2/^\circ\text{C}$ for a bond thickness of 0.002 in. Applying this material to the fins should not produce temperature gradients across the bond in excess of 0.5°C .

Surface finishes are listed in Table 13. The various solar-absorptance values were obtained by mixing white and black paint. It was found that the solar absorptivity of the resulting gray mixture has a linear relationship with the logarithm of the ratio of the carbon-black content of the black paint to the titanium-dioxide content of the white paint. A study is being conducted to obtain any required absorptivity from 0.2 to 0.9 by simply specifying the paint mixture ratio. If a desired revision of the cruise temperature field of the subsystem is indicated by the thermal model tests, it can be accomplished by simply refinishing the solar-illuminated areas with a higher or lower solar-absorptivity paint.

Table 13. Tentative thermal test model surface finishes and construction materials

Component	Surface	Material	Approximate thermal properties		
			Solar absorptivity	Emissivity at 100°C	Conductance, w/in. ² /°C
Thermal shield (conical and cylindrical surfaces)	Exterior	RCA 135 chromate film (Iridite)	0.35 → 0.40	0.06 → 0.08	—
	Interior	Cat-A-Lac black epoxy resin paint	—	0.9 → 0.95	—
Thermal fins	Solar side	Gray silicone ^a alkyd paint	Fin from top 1—0.59 2—0.61 3—0.57 4—0.63	≈0.9 Dependent on mixture	—
	Shadow side	White silicone alkyd paint	0.18	0.84	—
Camera temperature control plate and antenna ring	Solar side	Gray silicone ^a alkyd paint	0.27	≈0.9	—
	Interior side	Cat-A-Lac black epoxy resin paint	—	0.9 → 0.95	—
Structure and electronic modules	All exposed surfaces	Cat-A-Lac black epoxy resin paint	—	0.9 → 0.95	—
Interface between thermal shield and fins	—	Eccobond 45	—	—	≈2.5 ^b
Interface between structure and electronic modules	High dissipation	0.003-in. thick aluminum foil	—	—	≈1.1 ^c
	Low dissipation	0.003-in. thick aluminum foil	—	—	≈1.1

^aObtained by mixing white and black silicone alkyd paints. The shade of gray depends upon what value of solar absorptivity is desired.
^bFrom test data for 0.002-in. thick layer.
^cFrom test data.

3. Radiation Versus Conduction Coupling

A study was conducted to determine the influence of radiation coupling by elements of the subsystem as compared to conduction coupling. The thermal analysis model consisted of two concentric isotherms, the inner isotherm bounding an area of heat generation (equivalent to an electronic module having a circular mounting area placed in the center of a circular plate). The differential equation of heat transfer for the model was as follows:

$$\frac{d^2T}{dr^2} + \frac{1}{r} \left(\frac{dT}{dr} \right) - CT^4 + D = 0$$

where

r = radius

$$C = \frac{\sigma(\epsilon_A + \epsilon_B)}{Kt}$$

$$D = \frac{\epsilon_A E_A + \epsilon_B E_B}{Kt}$$

K = thermal conductivity

t = plate thickness

E = thermal energy striking the plate per unit area

and subscripts A and B refer to the two sides of the plate. The heat flow in the plate is given by:

$$Q_{(r)} = 2\pi(Ktr) \left(\frac{dT}{dr} \right)_r$$

The differential equation was solved on an analog computer for various values of C , D , and boundary conditions for temperature and temperature gradient at the outer isotherm radius. Over 2000 sets of temperature and temperature gradient curves were computed. A major conclusion was that radiation-coupling factors are of secondary importance when compared to conduction-coupling factors for the subsystem internal design. A decision was made to eliminate the lengthy digital-computer calculation of internal-configuration factors and radiation-coupling factors.

Since conduction is the primary mode of heat transfer within the subsystem, increased effort was placed upon analytical prediction of the conduction-coupling factors. Using conformal-mapping techniques, the conduction-coupling factor between two bodies on an infinite plate was determined. The solution is of the form: $R = Kt\psi$, the conduction-coupling factor, where

$$\psi = f(a) \frac{2(L/r_1)}{\left(1 + \frac{r_2}{r_1}\right) \ln \left[\frac{(L/r_1)^2 - \left(\frac{r_2}{r_1}\right)^2 - 1}{(r_2/r_1)} \right]}$$

$$f(a) = \ln \left[\frac{\left(\frac{L}{r_1}\right)^2 - \left(\frac{r_2}{r_1}\right)^2 - 1}{r_2/r_1} \right] \frac{1}{(T_1 - T_2)} \oint \frac{dt}{dn} \cdot ds$$

$$a = \left[(L/r_1)^2 - (r_2/r_1)^2 - \frac{(r_2/r_1)}{(L/r_1)^2 - (r_2/r_1)^2} - 1 \right] \left(\frac{r_1}{L} \right)$$

where r_1 and r_2 = the radius of the bodies, L the distance between body centers, and $f(a)$ was evaluated by graphical integration for $a = 2$. Values from $a = 1.1$ to 20 are being calculated on an IBM 709. A second analysis was developed to obtain the coupling factors for bodies on a finite plate. This program utilizes the results of the infinite-plate calculations. A report will be issued as soon as the data is available. The validity of the results will be checked by the thermal-vacuum tests on the test model.

An effort was made to estimate the most severe temperatures of the subsystem components using conduction coupling only; previous calculations assumed radiation coupling only. The assumption was made that the structure and some components (for example, the battery) were acting as heat sinks. The heat-sink temperature was evaluated by assuming that the total heat dissipation of the subsystem was transferred to the heat-sink components during the terminal-operation period. (None of the heat dissipation was considered to be lost by radiation from the shield.) Each of the components was then considered to be at terminal power for the operational-mode time, mounted to the high temperature heat sink. The energy dissipated in each component was assumed to be conducted to the sink through a heat path which has a length equal to half the height of the component and a cross sectional area equivalent to the circumference of the component base with an effective wall thickness of 0.060 in. The resulting temperatures for this extreme model are given in Table 14.

Joint conductance. A test was conducted on the various materials proposed for increasing the joint conductance between electronic components and the structure. A test device provided a known contact pressure on the joint material. The test was conducted at 100-psi joint pressure, this pressure corresponding to the lowest component mounting pressure expected for the subsystem. The test results are listed in Table 15. Lead foil gives the highest

value of conductance, but it will not be used for component mounting, since additional tests indicated that it could not withstand the shock and vibration environment. Thus, aluminum foil of approximately 0.003-in. thickness will be used on the thermal test model and evaluated.

Table 14. Estimate of component temperatures at impact

Component	Power ^a , w			T _R , °K	T _R , °C
	C	W	O		
Telemetry, A26	5	13.4	13.4	292	19
Power supply, A16 through 21	0	18.2	100	299	26
Transmitter, A14 through 19	0	0	92	316	43
Amplifier, A15 through 20	0	8	73	329	56
Four-port hybrid, A24	0	0	5	290	17
Dummy load, A25	0	0	60	305	32
Power relay, A17 through 22	0	14	14	300	27
Batteries, A10 and 11	1.25	100	200	295	22
Regulator, A12	9	42	44	311	38

^aC: cruise mode.
W: warmup mode.
O: operating mode.

Table 15. Joint conductance of various materials for 100-psi contact pressure

Material	Conductance, w/in. ² /°C
Silver foil (quilted)	1.6
Aluminum foil, 0.001-in.	1.2
Aluminum foil, 0.010-in.	1.0
Lead foil, 0.009-in.	2.8
Bare joint	0.6

G. System Integration

1. Electrical

The TV subsystem command switch has been simplified by using silicon-controlled rectifiers in place of the bulky high-current relays, and test console is being breadboarded for engineering-test-model-GSE integration. The test philosophy for checkout of the TV subsystem on the spacecraft bus has been finalized.

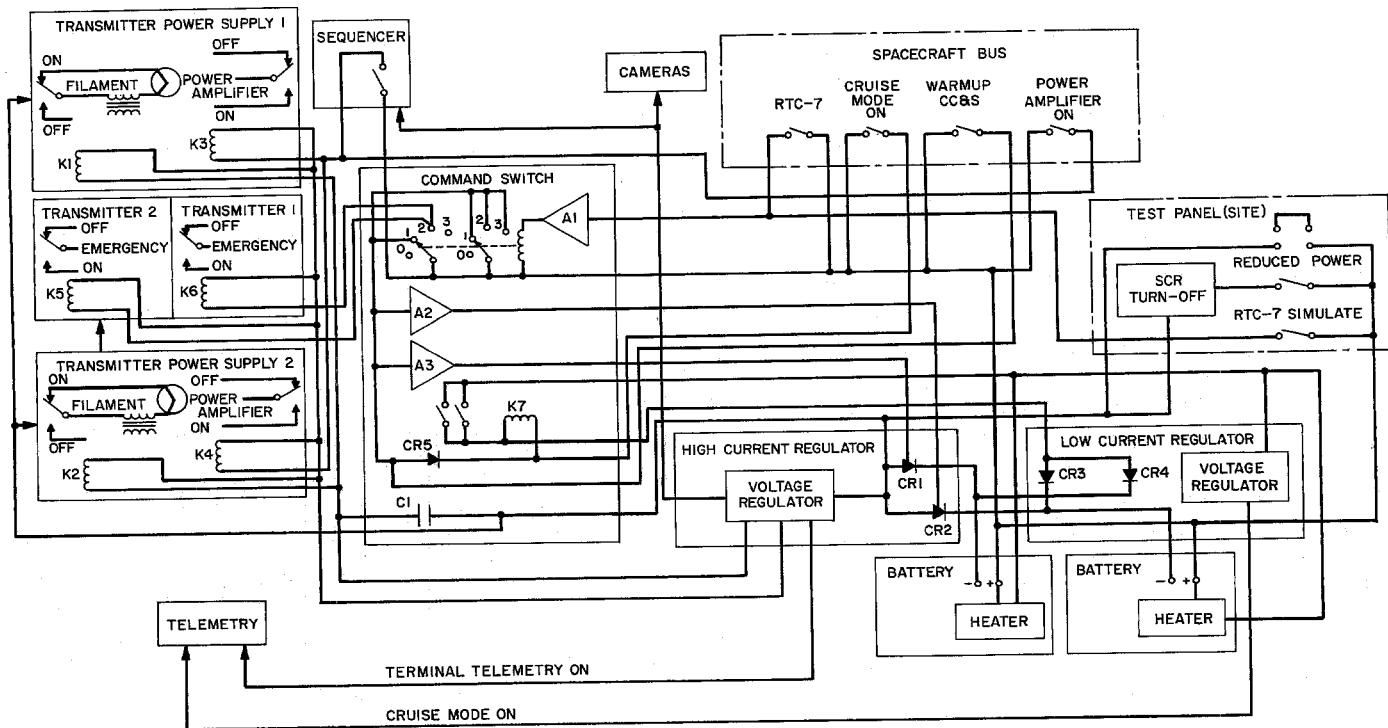


Figure 41. TV capsule control

Figure 41 shows the circuits and interconnections used for mode switching. The first signal received by the TV payload during flight is the *cruise-mode-on* signal, from central control and switching. This signal energizes Relay K7, which applies dc voltage to activate the voltage regulator. The -27.5-v output of the low-current voltage regulator starts processing and transmission of cruise-mode signals. During launch-pad checks the *cruise-mode-on* signal will not be available. Therefore, cruise-mode telemetry will not operate until warmup signal is given by the Real Time Command-7 (RTC-7) simulate signal. This signal energizes Amplifier A1, and the rotary solenoid switch moves from Position 0 to Position 1. The output of Position 1 is coupled through Diode CR5 to energize Relay K7. (Diode CR5 functions as a blocking device when the *cruise-mode-on* signal is received.)

The second signal is the *warmup* signal (prime source is RTC-7). When received, the rotary solenoid switch advances from Position 0 to Position 1. This turns on Amplifiers A2 and A3 sending gate turn-on signals to silicon-controlled rectifiers CR1 and CR2. The SCR's conduct, activating the high-current voltage regulator and transmitter power supplies. The high-current voltage regulator sends out -27.5 v to: (1) terminal telemetry, (2) camera electronics, (3) sequencer, and (4) command

switch. Voltage application to (1), (2), and (3) starts the planned and programmed operations of these assemblies. The voltage passes through the command switch to energize relays in other assemblies. A backup warmup signal from CC&S grounds the inputs of Amplifiers A2 and A3 directly to start the same chain of operation.

The third signal is the *full-power-on* signal from the CC&S. This signal energizes Relays K3 and K4 in the transmitter power supplies, providing proper voltages for full power-transmitter operation. This signal has a backup contact closure in the sequencer.

The fourth signal, *emergency telemetry command*, sets the TV capsule in the emergency mode of operation. When received, the rotary solenoid switch advances from Position 1 to Position 2 and Relays K5 and K6 are energized. Relay energizing disconnects the cameras from the transmitters, and connects the terminal telemetry signals directly to the transmitters. The warmup signal is still applied to Amplifiers A2 and A3 to ensure that both silicon-controlled rectifiers are conducting. (This is accomplished by the use of the second circuit on the switch wafer.)

The fifth signal is the *emergency-mode-off* signal. This signal advances the rotary solenoid switch to Position 3, de-energizing Relays K5 and K6. The cameras are again

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connected to the transmitters, and the TV subsystem performs in the normal operating mode. Position 3 is the final position to which the rotary solenoid switch will be advanced. On the launching pad, the next RTC-7 simulate signal will advance the switch to Position 0; the cycle of operating modes can be repeated if necessary.

A turn-off circuit for the silicon-controlled rectifiers is incorporated in the test console, since the rectifiers will continue conduction even after the gate signal is removed. Capacitor C1 is a part of the turn-off circuit, but is incorporated into the capsule because the turn-off circuit characteristics make it mandatory for the capacitor to be as close to the load as possible.

The reduced-power lead is grounded, causing Relays K1 and K2 to be energized when the plug of the hard line is mated with the jack. This prevents the transmitter from operating at full power on the launching pad.

2. Mechanical

TV-RA6-9
The mechanical test model has been fabricated, assembled, and vibration-tested without evidence of failure. The thermal model was fabricated and assembled, and delivered to test group for vacuum thermal testing. Assembly of the engineering test model has been started. All assembly fixtures and one shipping container have been received. Design is progressing on fixtures for supporting collimators for checking camera operation and sequencing. The mechanical test model is illustrated in Figures 42 and 43.

The general arrangement of components varies from the previous report only slightly. The designs of component mounting provisions have been completed and are currently being more precisely evaluated. The components are mounted in the structure by means of flanges or through bolting with the exception of the camera sequencer and camera electronics. These components are mounted using hold-down brackets.

A more precise weight schedule has been tabulated, using values obtained from engineering models and adopted changes. The new weights are presented in Table 16. The changes in weights and arrangements are relatively small, and variations in previously reported figures of center-of-gravity and moments are correspondingly small. A computer program has been completed that will enable changes and additions to be calculated more readily and accurately. The mechanical test model center-of-gravity and moment were calculated to be:

Center-of-gravity

$$Z = \text{Station 457.22}$$

$$X = -0.11 \text{ in.}$$

$$Y = -0.055 \text{ in.}$$

Moments, slug-ft²

$$I_{xx} = 19.78$$

$$I_{yy} = 20.30$$

$$I_{zz} = 3.50$$

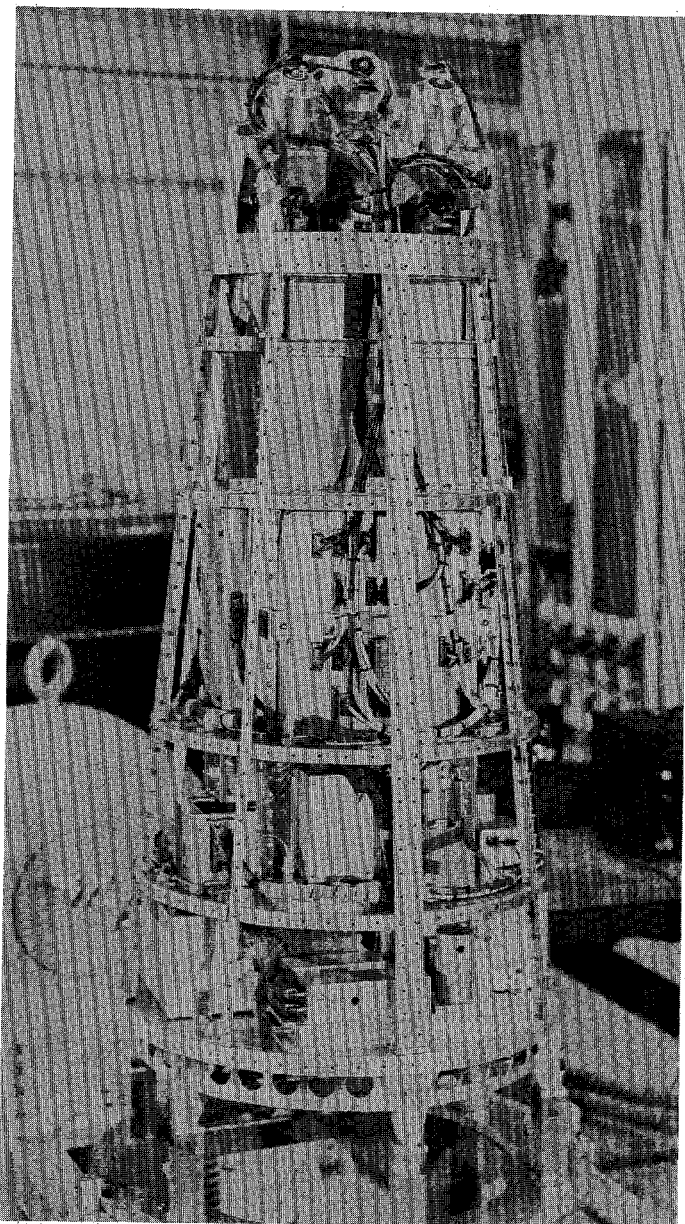


Figure 42. Mechanical test model - TV

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Table 16. Unit weights and dimensions

Reference designation	Unit	Weight, lb	Quantity
A1	Camera		
	Type-P	30.4	4
	Type-F	8.2	1
	Type-F and color wheel	10.1	1
	Bracket and hardware	6.5	1
A2 through A7	Camera electronics	43.8	6
A8	Video combiner	1.0	1
A9	Sequencer	9.0	1
A10 and A11	Batteries	87.0	2
A12	Regulator	3.0	1
A13	Command switch	2.0	1
A14 and A19	Transmitter modulator	8.0	2
A15 and A20	60-w amplifier	2.8	2
A16 and A21	Power supply	23.4	2
A17	Cruise mode regulator	0.75	1
A18 and A23	Signal samplers	0.4	2
A24	Four-port hybrid	1.25	1
A25	Dummy load	1.1	1
A26	Telemetry	10.7	1
A27	Temperature sensor	1.5	1
A28	Sequencer power supply	4.6	1
A29 and A30	Telemetry processing	0.8	2
	Structure	77.35	1
	Electrical harness	8.50	—
	Miscellaneous brackets	3.0	—
	Total weight	344.25	

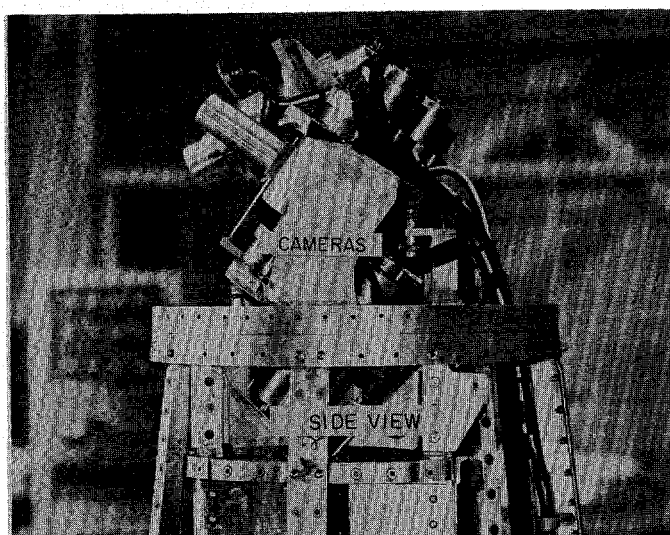
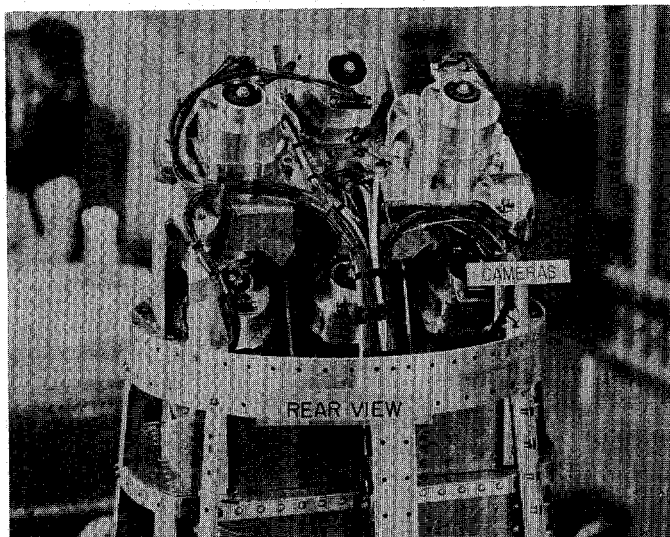
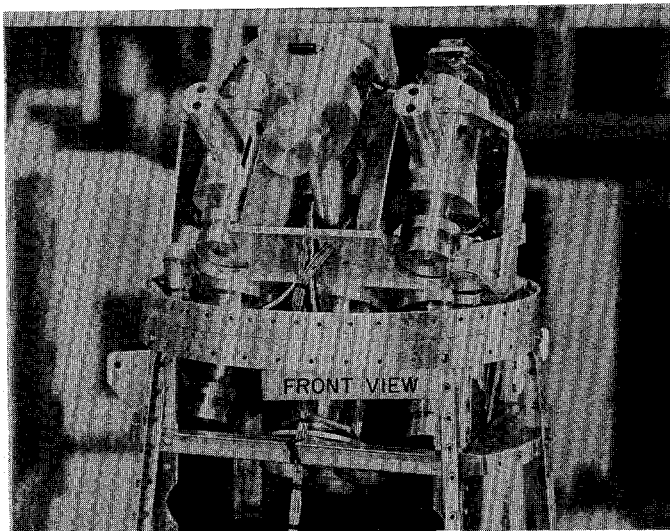


Figure 43. Mechanical test model, TV cameras

H. Ground Support Equipment

1. Data Recording and Display

Unit 1 is now in operation at AED, and is being used to train RCA service personnel. System tests of Unit 2 are scheduled to be completed at an early date.

Integration of the TV recording and display equipment with the ground communications equipment will begin when the ground test transmitter and the control panel have been received.

Dumont and Westinghouse kinescopes were received and are being evaluated in Unit 2. It is anticipated that both companies can supply tubes according to specifications.

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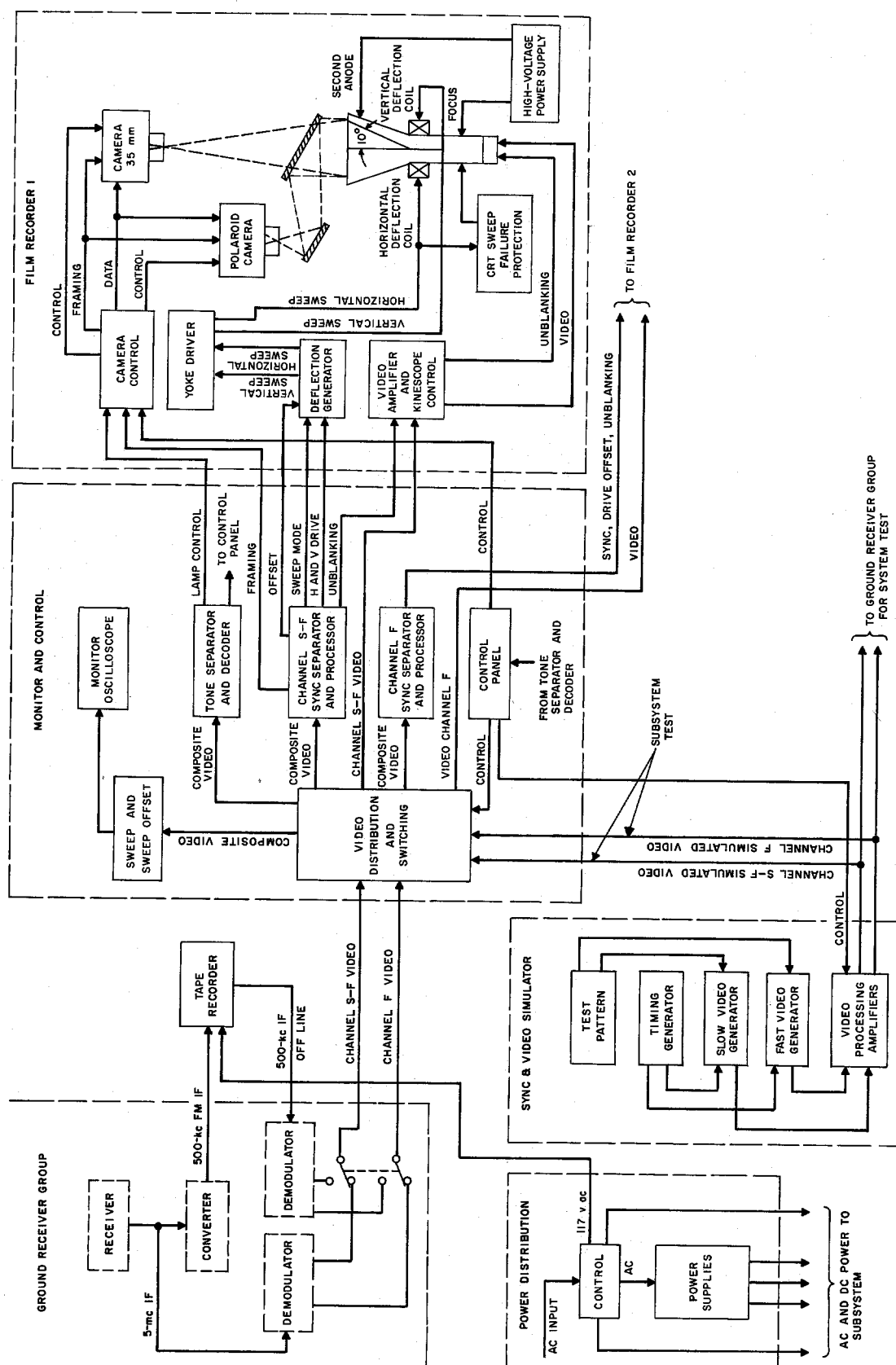


Figure 44. Ground TV recording and display equipment

Figure 44 is a block diagram of the ground recording and display equipment. The operational site configuration includes two tape recorders, two film recorders, one power distribution unit, one monitor and control unit, and one sync and video simulator. However, the checkout site configuration includes only one each of these units (Fig 45). In the operational site configuration, one film recorder records information from Cameras A and B until switch-over occurs; thereafter information from Cameras 1 through 4 is recorded. The other film recorder continually records information from Cameras 1 through 4. Camera A-B pictures are displayed alternately by the film recorder kinescope, with the Camera A display including an indicator light position on the 35-mm camera data board to indicate the color filter used by the TV payload camera. Three filter colors (red, green, and blue) and neutral are utilized. The Camera B display immediately following the Camera A display does not have a color-filter-identification indicator.

The breadboard tape recorder demodulator has been improved by a careful alinement and slight modifications in the filter matching circuitry. These changes have reduced transient problems to an acceptable value, but have resulted in lowering the video bandwidth from 200 kc to -3 db at 150 kc. The final model has an improved output amplifier and filter circuit which should prove more desirable.

2. Communications

Unit 1 was received in January by AED for system evaluation with the TV recording and display equipment. Unit 2 will be shipped to JPL in March (Fig 46).

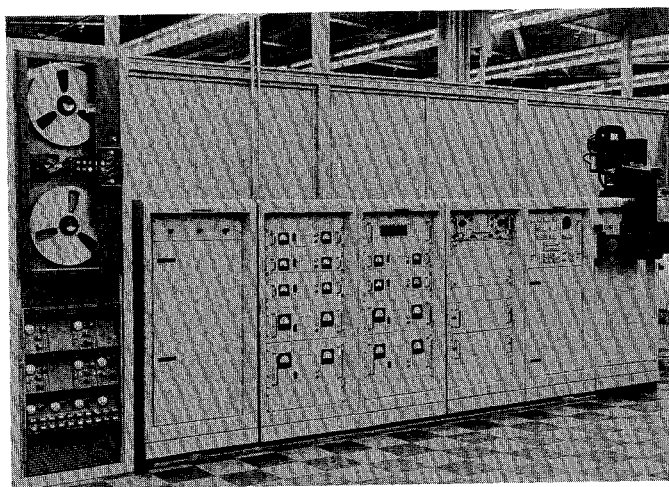


Figure 45. Ground equipment, TV data recording and display

Several receiver modifications have been made, including filters to reduce interchannel interference. New control panels, signal simulator, and a new test transmitter will be delivered to AED for final system integration at an early date.

Control panel functions are listed in Table 17.

The redesign also added functions that are readily accessible to the strip recorder. Any one of the following can be selected:

- (1) Channel 8 demodulator.
- (2) Telemetry (T/M) emergency data Channel A.

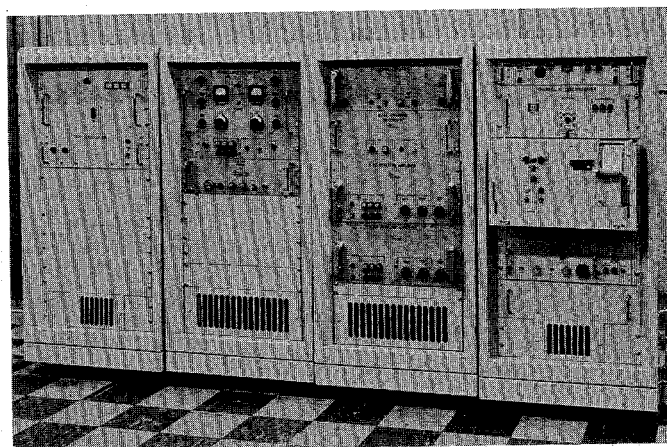


Figure 46. Ground receiving equipment

Table 17. Control panel functions

Switch	Function
Circuit breaker	Primary power
Channel 8 demodulator input	Either telemetry simulator or payload
Pre-emphasis input	Either full scan or partial scan simulated video
Test transmitter input	Either tape demodulator or out of pre-emphasis for simulated video
225-kc telemetry simulator	Either to test transmitter or 225-kc demodulators
Telemetry emergency control	On-off
Test transmitter channel selector	Either Channel A or Channel B
Channel A video telemetry input	Either 0.5-mc detector-amplifier or tape demodulator
Channel B video telemetry input	Repeat as per Channel A
Telemetry input, Channel A	Either 225-kc telemetry simulator or capsule input
Telemetry input, Channel B	Either 225-kc telemetry simulator or capsule input
Tape demodulator input: record tape playback	Either Channel A or B

(3) T/M emergency data Channel B.

(4) T/M 225-kc data Channel A.

(5) T/M 225-kc data Channel B.

The telemetry simulator unit will have the capability of providing either the 225-kc T/M rate or the emergency 3-kc rate. The voltage controlled oscillator (VCO) inputs or the emergency output is representative of a commutator. Four data voltage levels can be selected: 0, -1, -3, -5.

Some changes in the over-all bandpass characteristics of the tape demodulator, particularly at 200 kc, are being investigated. However, since part of its function is tied in with portions of the over-all system requirements, various trade-offs between units are being considered. Aspects of concern are demodulator bandpass, low-pass filter, and inversion of one of the video channels.

I. Testing

1. Test Philosophy of Spacecraft-Mounted TV Subsystem

a. Description and preliminary conditions. While the TV subsystem is mounted on the bus, the test access will be available through the TV subsystem test connector and through the hard line access provided by JPL in the junction box. The test connector and hard line access will be used to completely exercise the operating functions in the TV subsystem. The 960-mc RF output from the payload and the 3-kc output used for the cruise-mode telemetry will be cable-connected from the bus communications equipment to the ground support equipment.

A power-control relay assembly, which switches between the payload batteries and the external power supply, and also allows battery charging and monitoring, will be mounted externally on the TV subsystem during preliminary system tests. This assembly, which is not a part of the TV subsystem, is connected between the power supply rack and the TV subsystem. A battery-monitoring cable will be connected from the batteries to the power supply rack to provide continuous monitor-

ing of the battery operating temperature and individual battery cell voltages.

The test connector will contain:

- (1) An output from each of the three video combiner amplifiers.
- (2) An output from each of the two 225-kc VCO's used for the terminal-telemetry mode.
- (3) Outputs for monitoring each of the four command switch positions.
- (4) Outputs to control the operation of the sequencer.
- (5) A dc-return output used in the switch controls at the test console.

The outputs of the test connector will be directly connected to a video line driver assembly, which mounts externally and is not part of the payload. The outputs from the combined three video amplifiers are summed by an amplifier in the video line driver assembly. This amplifier also de-emphasizes, inverts, and amplifies the video signal. The type of test video desired can be selected at the test console.

The output of the video amplifier assembly is routed to the subsystem ground equipment by four cables:

- (1) Two coax cables containing the 90-point test telemetry will be connected into the RCA ground equipment receiver.
- (2) One coax cable containing the summed video will be connected to TV data recording and display equipment.
- (3) One cable containing the control lines to the sequencer, command switch, and switch and power inputs to the video amplifier will be connected to the test console.

The hard line access provided at the bus junction box will contain battery monitoring and charging lines, a real time command simulate line, a system turn-off line, and a control line to inhibit full power operation on the launch pad.

The following ground checkout equipment will be available and will be used during the tests of the TV

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subsystem: (1) TV data recording and display equipment, (2) ground receiving equipment, (3) test console, and (4) power supply and monitoring equipment.

b. Initial requirements. Before mounting the TV subsystem to the bus, the externally mounted power-control relay assembly must be electrically connected between the batteries and the regulator. The cables from the power supply rack must then be connected to the power-control relay assembly.

After the TV subsystem is mechanically mounted onto the JPL bus and the interface connectors are connected, the hard line wires from the TV subsystem must be checked out for continuity. The continuity check can be made at the power supply rack and at the test console cable interface plugs. The continuity path, which is a check of the eight hard line wires, will extend from the TV subsystem through the subsystem-bus interface connector, through the bus ring harness, through the hard line cable, and through the bus junction box to the subsystem ground equipment.

Once the hard line wires are checked, the circuit breaker of the power supply rack may be closed. The test director can select either battery power or external power at the power supply rack. Voltage readings will be made at the power supply rack.

c. Test procedure. After completing initial requirements for the power turn-on, the test director will place the command switch in the zero position by depressing the RTC-7 simulate pushbutton on the test console until the monitor light of the command switch position indicates *zero*.

The test director can now place the TV subsystem into the warmup mode by depressing the RTC-7 simulate pushbutton once. The command switch position monitor light will switch from *zero* to *warmup*. Channel 8 telemetry information will be transmitted by the JPL 3-w transmitter to JPL ground communications.

Channel 8 telemetry modulated signal will be provided to the subsystem ground equipment by a cable link. The cruise-mode telemetry will be recorded on the strip recorder and compared to a previously prepared master telemetry chart. The cruise-mode telemetry is the only transmitted information available from the TV subsystem during the warmup mode. The cruise-mode telemetry will be transmitted during all of the TV subsystem

modes. The same test will be performed with the activation of the cruise-mode CC&S command with the command switch position in the zero position. Once the cruise-mode CC&S command is activated, the same cruise-mode telemetry will be transmitted. Since the cruise mode will already have been checked in the previous test, this test need not be as extensive. Fundamentally, the test will be performed to ensure that the pair of wires from the CC&S cruise-mode contact are correctly interfaced to the TV subsystem.

If the cruise-mode telemetry is turned on by the CC&S command, the test director must depress the RTC-7 simulate pushbutton on the test console to place the TV subsystem into warmup. If the cruise-mode telemetry is turned on by depressing the RTC-7 simulate pushbutton once when the command switch position monitor is lighted *zero*, then no further command need be initiated by the test director since the TV subsystem will already be in the warmup mode. The command switch position monitor warmup light will be lighted.

The warmup mode turns on the high-current regulator, thereby distributing voltage to the entire TV subsystem. The following events occur at the warmup command:

- (1) The sequencer starts its timing operation with the 5-min warmup period and also starts its sync-pulse generation.
- (2) The cameras, camera electronics, and shutters start opening.
- (3) The 90-point telemetry is activated.
- (4) The communications equipment is in warmup.

Only the cruise-mode telemetry continues to be transmitted and recorded during the warmup mode. The test outputs of the video and 90-point telemetry will be available at the test connector during the warmup cycle. The 90-point telemetry 225-kc test output will be demodulated and recorded on the strip recorder. It can also be checked in the same manner as the cruise-mode telemetry, by using a previously prepared master telemetry chart for the 90-point telemetry.

A six-collimator array will be used to illuminate the camera vidicons to provide video-output signals. The test director will select the type of video signal by depressing a button in the test video select group on the test console. The composite video-test output signal from the TV subsystem will be de-emphasized, inverted, and

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amplified in the line driver video amplifier assembly mounted externally to the payload. This signal is then routed to the data recording system for processing, displaying, and checking.

Normally, the test outputs for both the video and 90-point telemetry signals will not be used; however, these outputs are available once the warmup mode is turned on. Preliminary checks can be made without putting the TV subsystem into the full-power-on mode. The normal data channel in the full-power-on mode will be via the RF link, which is from the RF connector at the bus coupler to the subsystem ground equipment.

The video and telemetry signals will be transmitted after the sequencer has counted the 5-min for the warmup mode and has provided a full-power-on command. Through sequencer control switches, provided on the test console, the test director may elect to stay in the warmup mode until all of the engineering measurements are taken and verified. If these controls were not provided, the sequencer would automatically place the TV subsystem into full-power-on.

The test director can manually place the TV subsystem in the full-power-on mode by depressing the transmitter-power pushbutton on the test console, or he can wait for the sequencer to count 5 min after the warmup command is initiated. The CC&S full-power-on contact also will be tested after the test director inhibits the sequencer full-power-on command. He will then request activation of the CC&S full-power-on contact. In the full-power-on mode, the transmitter power amplifiers are turned on, and normal transmission of video and telemetry signals will start. The high power operation of the transmitters will be interlocked when the subsystem-bus interface connector is connected, preventing the transmission of the full 60-w output of the high power amplifier. Provisions will be made to test the 60-w output in one of the steps of the test specification. It is important that the full-power-test pushbutton on the test console be pushed only in the warmup mode to prevent damage to the transmitter tubes. Power and frequency measurements will be made throughout the full-power-on mode.

During the full-power-on mode, the vidicon lenses of the six cameras will be illuminated by the collimator array. The video signal may be analyzed in the ground equipment against the specifications. The 90-point telemetry also will be transmitted during the full-power-on mode as part of the RF output from the TV subsystem. The telemetry will be separated from the video, be

processed, and then recorded in the ground receiving equipment for check against the master telemetry chart.

Another full-power-on mode check will be to test the output of Camera 1 which can free-run in the event that the sequencer clock pulses are absent. The test director will depress the sync-disable pushbutton on the test console. Only the output of Camera 1 will be transmitted and processed by the TV data recording equipment to determine whether the free-running camera sync generator is operating correctly.

The test director will check the emergency telemetry operation while the TV subsystem is in the full-power-on mode. He will depress the real time command simulate pushbutton on the test console or request that activation of the RTC-7 contact to step the command switch to the emergency position. The command switch is not used to turn on the full power; only the sequencer and the full-power-on CC&S contact can activate the full power operation. Thus, the command switch position monitor light on the test console will have remained lighted in the warmup position. After the test director gives the emergency command, the position monitor light will switch from warmup to emergency. The outputs of the three summing video amplifiers in the TV subsystem will now be switched, and the 90-point telemetry commutating switch will now directly modulate the transmitter. The emergency telemetry will be received, processed, and recorded by the ground receiving equipment to check for proper operation of the emergency telemetry. Upon completing the emergency-mode test, the test director will depress the real time command simulate pushbutton once again or request the RTC-7 contact. The emergency mode will be switched off with the video outputs again transmitted to the ground equipment. The command switch position monitor will be lighted *emergency-off* on the test console. To turn off the TV subsystem, the test director will depress the RTC-7 simulate pushbutton on the test console until the zero switch position monitor light is on. The test director also will request that JPL release all the CC&S contacts. Once the two steps are completed, the test director will depress the system turn-off pushbutton.

2. Environmental Testing

During the period since December 15, 1961, the preparation and issuance of the test plan and necessary specifications have been nearly completed, the specific arrangements for testing have been made, and testing has begun.

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Environmental documents prepared or being prepared are as follows:

Document	Status
<i>B/k III</i> Environmental Test Plan for the Ranger TV Subsystem	Issued January 18, 1962
Assembly Level Environmental Specification for the Ranger TV Subsystem	Revised and reissued reissued December 16, 1961
Specification for the Mechanical Test Model (MTM) of the Ranger TV Subsystem	Issued January 10, 1962
Specification for the Thermal Test Model (TTM) of the Ranger TV Subsystem	Issued
Specification for the Proof Test Model (PTM) of the Ranger TV Subsystem	In preparation
Specification for the Flight Model of the Ranger TV Subsystem	In preparation

Test fixture design is complete, and the remaining necessary procurement procedures have begun. A number of fixtures were delivered and used in actual testing.

Qualification (type approval) testing of assemblies began with the receipt of the battery in mid-January. The battery has successfully completed all tests except the ethylene-oxide sterilization test (scheduled for an early date). The temperature sensor and the several components of the communications group are also under tests.

The mechanical test model was subjected to vibration testing on the 28,000-lb shaker in accordance with requirements, and successfully completed all testing.

B/k III
A decision not to purchase an ethylene-oxide spacecraft sterilization system required that other arrangements be made in order to completely qualify the assemblies. Therefore, after investigating the availability of facilities which are reasonably accessible for this purpose, satisfactory arrangements have been made with the U.S. Army Chemical Corps Biological Laboratories, Fort Detrick, Maryland. It is planned to minimize by scheduling the number of trips necessary to qualify the entire group of assemblies without unduly causing delays in the test program.

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IV. Surveyor Project

Surveyor is a lunar-surface-investigation project which includes seven missions for soft landing modest scientific payloads on the lunar surface in 1963, 1964, and 1965. The project is managed by the Jet Propulsion Laboratory for the National Aeronautics and Space Administration. Hughes Aircraft Company is under contract to JPL to develop and operate the spacecraft.

The launch vehicle system will be the *Atlas-Centaur* involving the cooperation of Marshall Space Flight Center at Huntsville and its Launch Operations Directorate at Atlantic Missile Range. Tracking and spacecraft control will involve the cooperative use of the JPL Deep Space Instrumentation Facility and the Space Flight Operations Facility.

The performance objectives of the *Atlas-Centaur* launch vehicle system have been under study for the past several months by the NASA Headquarters and the Marshall Space Flight Center. It is anticipated that the objectives will be established during the next reporting period.

While the *Atlas-Centaur* performance objectives are being established, the spacecraft development is continuing assuming the originally estimated gross separated weight of 2500 lb. However, to insure compatibility with a possibly lower performance launch vehicle system, studies have been undertaken to determine the feasibility of developing a lightweight *Surveyor* spacecraft. Details of these studies are reported in Section V.

The following material relating to the *Surveyor* Project is reprinted directly from the Hughes Aircraft Company report: *Surveyor Spacecraft System, Bimonthly Progress Summary*, March 1962.

In December 1961 the *Surveyor* spacecraft design was placed under formal change control. Since that time, only a few design modifications have been incorporated in the basic bus and scientific instrument payload integration.

Nearly all of the instrument payload has reached the prototype hardware phase of development. Most of the television circuitry is in package design: improved photosurfaces for the vidicons are being investigated, and provisions have been added for thermal control of the vidicon faceplates. The basic design of the tape recorder has been completed. Developmental testing of the television mirror assembly, instrument deployment mechanisms, subsurface sampler, and soil sample processing system is continuing.

Further analysis has been performed to optimize the mid-course correction logic and maximize vernier propulsion fuel margin. Other system analysis has included definition of the required acquisition region for the doppler radar under 45-deg approach conditions. Optimization of the coast phase attitude and vernier descent phase control systems is continuing.

In flight control development, an improved version of the inertial reference unit has completed preliminary

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design. Sun sensor tests are continuing and tests of the developmental model star sensor are about 20% complete.

Many of the electronic subsystems, such as the control signal processor and the command receiver-transponder interconnections, are in the package design phase. The electrical prototype of the planar array antenna has met design requirements in tests to date, demonstrating a gain of 26.9 db. Two new versions of the omnidirectional antenna are being built for test. Circuit design for the command decoding system is nearly completed. Satisfactory patterns are being obtained with the altitude marking radar antenna. In the electrical power system, the first developmental solar panel performed satisfactorily in tests at Table Mountain. Tests of experimental battery cells indicated that the design requirements can be met. Following temperature testing of the optimum charge regulator and boost regulator, final design release of the power control circuitry will be made.

An engineering-type, full-size mockup of the flight-type spacecraft has been built. Construction of the T-1 landing dynamics test vehicle has been completed. An improved extension system for the omnidirectional antennas has been designed; it will give increased coverage and simplify coaxial cable deployment. Landing gear tests are being made, and the drop test facility for the T-1

vehicle has been checked out, ready for drop tests in late February.

A refined thermal analysis of the temperature-controlled compartments, with radiation interchange effects from all spacecraft surfaces taken into account, has been prepared for a computer study. Preliminary thermal analysis of most of the scientific instruments has been accomplished. Review of the instrument structural design indicates that further stress analyses of these items should be made.

The flight-weight main retro-rocket case has successfully completed hydrostatic pressure tests. A propellant variation with improved physical properties will be tested in an attempt to eliminate propellant cracking problems encountered in the retro-rocket. The initial tests of the complete vernier propulsion system were satisfactory.

An updated estimate indicates a mean spacecraft system reliability of 0.75. Detailed plans for conduct of the T-2 flight control system tests at AFMDC are being formulated, and a detailed specification for conducting proof-of-design tests with the T-4 spacecraft is being prepared. A preliminary AMR Pad Safety Report was completed. The wide-band portion of the demodulator for the command and data handling console has been released for package design, and the IF circuit design will be released soon.

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A. System Design Summary

1. System Description

The spacecraft design was placed under formal change control on December 22, 1961. The spacecraft configurations for the stowed, midcourse, and postlanding conditions are shown in Figure 1. Only the small controlled changes described in Section J have been made in the last report period.

A significant design decision has been made in the main retro-rocket engine propellant. In order to improve the low-temperature characteristics of the rocket engine, the propellant formulation (in terms of physical properties, only) has been changed from the HA series PBAA to the HC series. The new, HC, propellant will allow a broader operating temperature range, and should eliminate the low-temperature cracking previously experienced.

A detailed sequencing study of electrical power during the lunar descent phase has shown the 150 w peak capability of the boost regulator to be inadequate. The boost regulator has been redesigned to provide a peak capability of 175 w; at the same time, the low-load efficiency of the regulator has been improved.

The spacecraft functional block diagram for the first two flights (SC-1, SC-2, SC-3) is shown in Figure 2. The block diagram has been brought up to date and expanded to show individual engineering measurement accelerometers, individual pin pullers, and umbilical ties between

the spacecraft and the GDA-supplied *Centaur*-to-spacecraft adapter. The block diagram is under formal design control, and is revised only under the surveillance of engineering change management.

2. Weight Summary

Current weight status is summarized in Table 1 and shown in detail in Table 2. The predicted flight weight of the scientific payload has decreased 9.2 lb from that last reported, but the predicted flight weight of the basic bus has increased 20.6 lb. This increase has been primarily in the electronics of the flight control sensor group, where the number of transformers and other circuit components has increased, and in the spacecraft vehicle equipment compartments, due to thermal control requirements and increased volume.

Table 1. Spacecraft weight summary

Item	Maximum allowable weight, lb	Current design weight, lb	Predicted flight weight, lb
Total separated weight	2,500.00	2,572.8	2,550.0
Basic bus	553.0	639.4	617.4
Usable propellant	1,602.0	1,591.0	1,591.0
Scientific payload	345.0	342.4	341.6

3. Electrical Energy

An electrical energy summary is shown in Table 3. This summary assumes launch on the eighth day of the launch window in August, 1963; for this launch condition, solar energy will be available to the spacecraft for 163 hr from touchdown until crossing of the first day/night terminator. The current estimated energy requirements are based upon the payload specified for Missions P-42 and P-43.

The 5926 w-hr estimated daytime energy requirement of the scientific payload results from the assumption that all daytime experiments will be completed on the first lunar day. Since this assumption leads to a reduction of the daytime energy reserve to approximately one-half the allowable reserve, the mission sequence for the eighth day of the launch window must be revised to defer a portion of the payload daytime operation to the second lunar day.

Table 2. Weight breakdown

Item	Maximum allowable weight, lb	Current design weight, lb	Predicted flight weight, lb
1.0 BASIC BUS (total)	553.0	639.4	617.4
1.1 Flight control		51.6	51.4
Sensor group			
Secondary solar sensor			
Attitude control system			
1.2 Electronics		93.9	93.9
Data link (antennas, RF switches, transmitter-diplexer, command receiver and transponder)			
Central command decoder			
Central signal processor			
Altitude marking radar			
Doppler velocity sensor and altimeter			
Power control system			
TV camera 4			
Engineering data processing			
Mechanisms auxiliaries			
1.3 Electrical power		38.5	38.5
Solar panel			
Batteries			
1.4 Mechanisms		32.0	28.4
Positioner, antenna/solar panel			
Antenna mechanisms, omnidirectional			
Safety and arming provisions			
Separation sensing device			
1.5 Spacecraft vehicle		203.0	195.0
Spacecraft structure			
Landing gear installation			
Equipment attaching hardware			
Equipment compartments A, B, and C			
Electrical wiring			
Pneumatic lines, attitude control			
Release mechanism, retro-rocket			
Engineering measurements sensors			
1.6 Spacecraft propulsion (not including usable propellant)		220.4	210.2
Rocket engine, main retro			
Vernier propulsion system			

Item	Maximum allowable weight, lb	Current design weight, lb	Predicted flight weight, lb
1.6 (continued)			
Unusable vernier propellant			
Helium			
2.0 USABLE PROPELLANT (total)	1,602.0	1,591.0	1,591.0
2.1 Retro-rocket propellant		1,430.4	1,430.4
2.2 Vernier propellant		160.0	160.0
2.3 Pyrogen propellant (igniter)		0.6	0.6
3.0 SCIENTIFIC PAYLOAD (Missions P-42 and P-43) (total)	345.0	342.4	341.6
3.1 TV Cameras 1, 2, 3 and mirrors		45.3	45.3
3.2 High-resolution telescope and camera		15.9	15.9
3.3 Subsurface sampler		43.5	43.0
3.4 Surface sampler		9.0	9.0
3.5 Spacecraft anchoring device		10.3	10.3
3.6 Sample processor		13.5	13.5
3.7 X-ray spectrometer		28.0	28.0
3.8 X-ray diffractometer		20.5	20.5
3.9 Gas chromatograph		12.0	12.0
3.10 Surface geophysical subsystem		31.7	31.6
Surface density instrument and acoustic sensor			
Surface thermal diffusivity instrument			
Surface magnetic susceptibility instrument			
Acoustic sensor			
Acoustic source			
Soil mechanics instruments			
Surface temperature instrument			
Penetrometers			
Surface geophysical auxiliaries			
3.11 Subsurface logging sonde		5.2	5.2
3.12 Tape recorder		11.8	11.8
3.13 Ionization chamber assembly		5.0	4.8
3.14 Scientific payload batteries		62.0	62.0
3.15 Electrical wiring		17.0	17.0
3.16 Geophysical thermal and TV auxiliaries		4.1	4.1
3.17 X-ray and radiation detection auxiliaries		2.8	2.8
3.18 Surface sampler, gas chromatograph, and acoustic velocity auxiliaries		4.8	4.8

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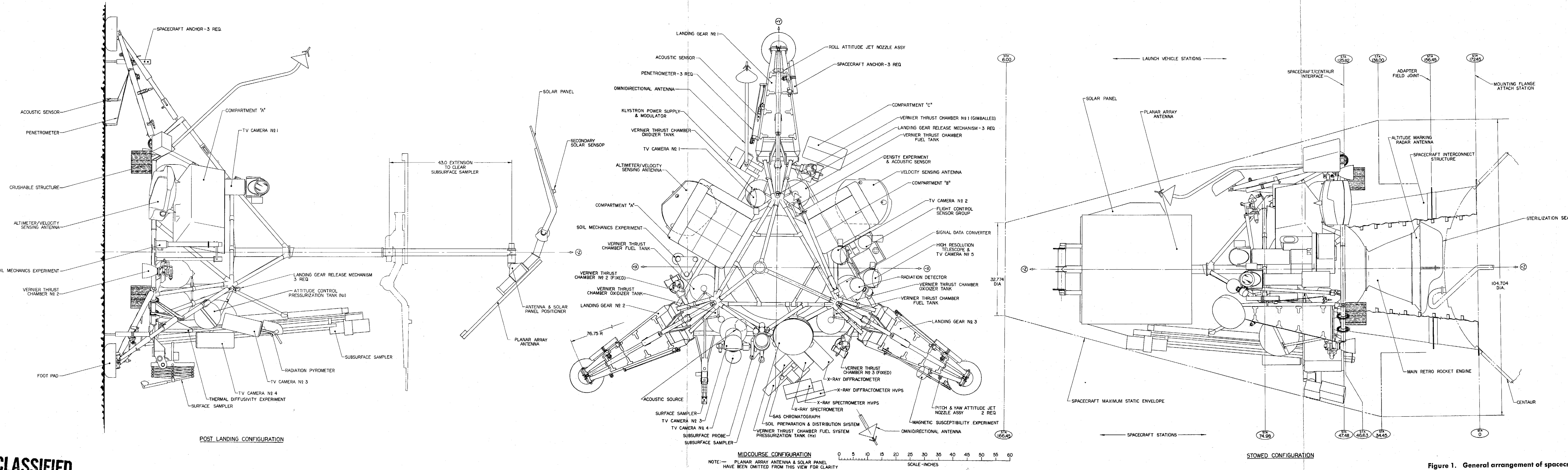


Figure 1. General arrangement of spacecraft

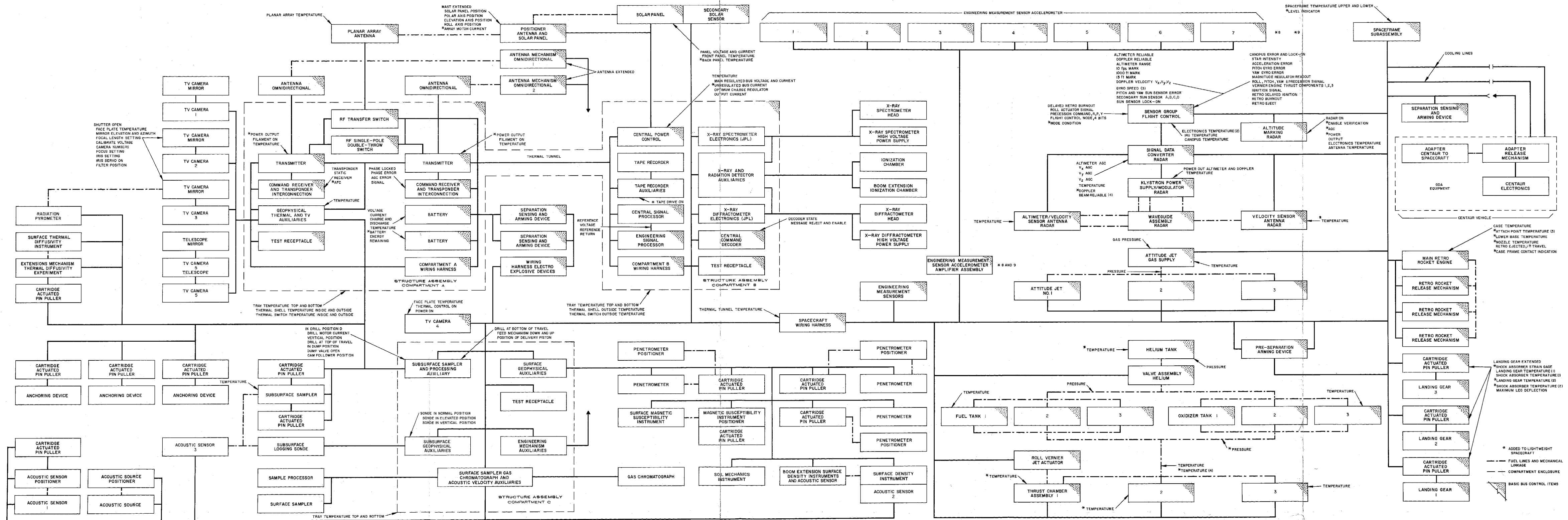


Figure 2. Spacecraft functional block diagram

Table 3. Electrical system energy summary

Energy	Daytime, 163 hr		Nighttime, 350 hr	
	Allowable watt-hours	Current estimated watt-hours	Allowable watt-hours	Current estimated watt-hours
Total available energy	7628	7628	5300	5300
Scientific payload operation	5355	5926	4128	4128
Basic bus operation	745	884	827	886
Energy reserve	1528	818	345	286

4. Command and Data Channels

Command and data channels now being provided in the spacecraft are summarized in Tables 4 and 5. Command and data channels listed for the scientific payload include those required by the Hughes-supplied instrument auxiliaries, both mechanisms and electronics, in addition to the specific instrument requirements. In Table 5 the data channels are divided into the four methods used to telemeter the data. The figures indicated in this summary are based upon the scientific payload specified for Missions P-42 and P-43.

5. Lightweight Design Summary

During the last report period, at the request of JPL, Hughes Aircraft Company made a detailed study of the feasibility, developed a configuration, and established a development plan for a "lightweight" spacecraft which could be used during the early part of the *Surveyor* program. The results of the Hughes efforts were reported in "Study of Lightweight Surveyor-A" (SSD232P), dated February 1, 1962.

The lightweight design (Fig 3) was made feasible primarily by a reduction in the weight of scientific payload to be carried and a corresponding reduction in usable propellant. The basic objectives and ground rules used in arriving at the lightweight spacecraft design are outlined below.

Objectives

(1) Maximize the probability of a successful soft landing on the lunar surface. Verify achievement of the soft landing and evaluate the engineering performance of the spacecraft design.

(2) Within the limitations of payload constraints, determine the lunar environment and selenographic characteristics.

(3) Maximize the probability of lunar night survival.

Table 4. Spacecraft commands

Basic bus	
Data link and TV Camera 4	29
Signal processing	31
Mechanisms and propulsion	27
Mechanisms and vehicle	20
Electrical power and radar	19
Flight Control	28
	154
Scientific payload	
Tape recorder	13
Surface geophysical	28
Subsurface geophysical	8
Geophysical thermal	14
Acoustic velocity	9
Subsurface sampler and soil processor	25
Surface sampler	8
Gas chromatograph	12
X-ray diffractometer	11
X-ray spectrometer	13
Radiation detector	9
TV subsystem	32
	181
TOTAL	335

Table 5. Spacecraft data channels

Item	Commuted data (PCM)		High-frequency or continuous data (PM or FM)	
	Digital	Analog	Subcarrier oscillators	Main carrier
Basic bus				
Data link	7	8	1	—
TV Camera 4	2	2	—	1
Signal processing	—	2	—	—
Mechanisms	4	5	—	—
Propulsion	—	9	—	—
Vehicle	4	20	4	—
Electrical power	—	22	—	—
Radar	1	13	—	—
Flight control	36	32	1	—
	54	113	6	1
Scientific payload				
Tape recorder	—	—	7	—
Surface geophysical	1	12	5	—
Subsurface geophysical	—	3	3	—
Geophysical thermal	2	10	—	—
Acoustic velocity	1	—	5	—
Subsurface sampler and soil processor	12	7	—	—
Surface sampler	—	4	—	—
Gas chromatograph	2	10	—	—
X-ray diffractometer	—	2	—	2
X-ray spectrometer	—	3	1	1
Radiation detector	2	2	1	—
TV subsystem	2	10	—	1
	22	63	22	4
TOTAL	76	176	28	5

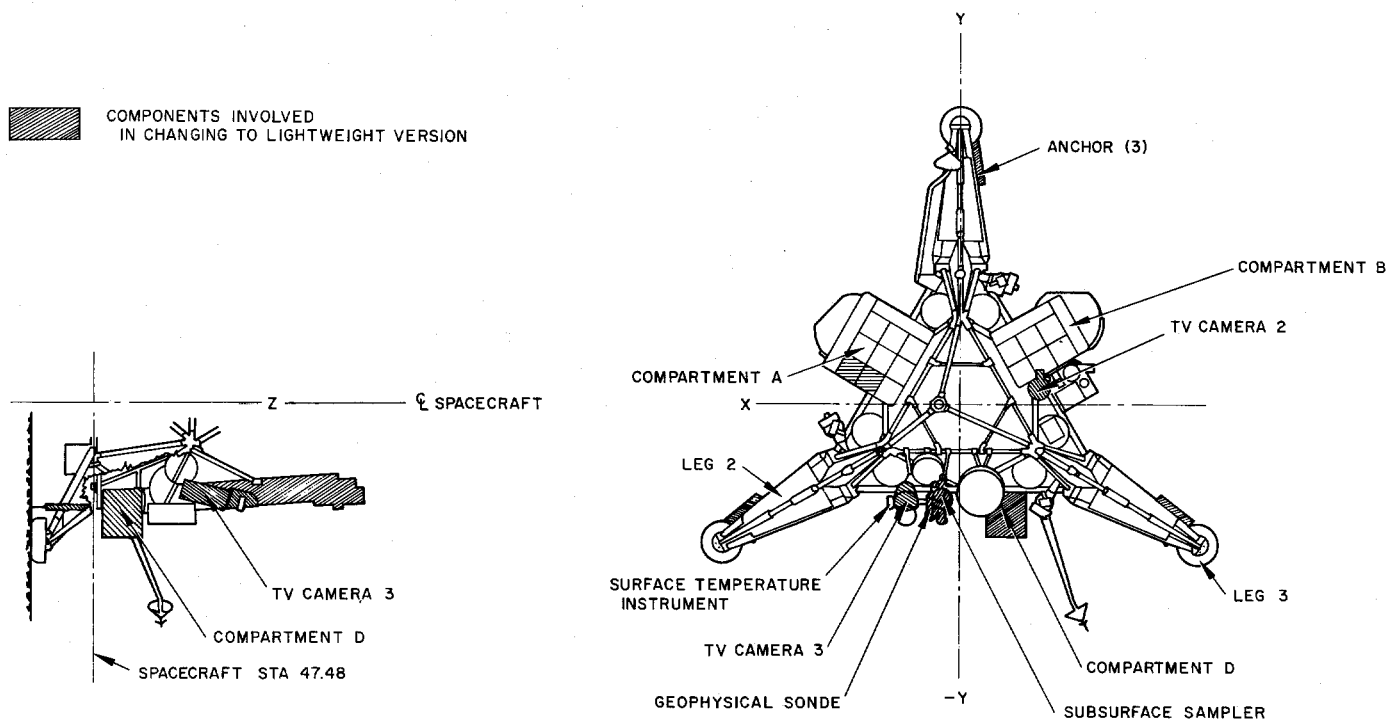


Figure 3. Configuration of 2100-lb spacecraft

Ground Rules

(1) The requirements of the JPL Specification 30240A shall apply to the lightweight spacecraft.

(2) The payload should include the surface surveillance TV, subsurface sampler, and subsurface geophysical probe.

(3) The total spacecraft shall be as light as possible consistent with engineering and scientific requirements. However, the total weight at *Centaur*/spacecraft separation shall not exceed 2100 lb.

(4) Predicted flight weights shall be used in establishing the weight of the spacecraft for this study.

(5) There shall be no change in the P-42 launch date. The lightweight spacecraft shall be developed on schedule consistent with the August 1963 launch date.

(6) No new major components shall be developed which are peculiar to the lightweight vehicle design. There will be no changes to basic bus component designs which would make extensive requalification necessary.

(7) The lightweight spacecraft design shall be fundamentally the same as the 2500-lb spacecraft. The only major difference between the two spacecraft will be in the amount of payload and expendable fuel carried.

(8) There will be no compromise in the completeness or thoroughness of the type approval or flight acceptance test programs in developing the lightweight spacecraft.

(9) Additional engineering instrumentation will be added to the spacecraft to assist in evaluating and verifying spacecraft performance.

One of the primary factors influencing the configuration of the lightweight spacecraft was that of arriving at an acceptable lateral center of gravity location. Since the position of the main retro-rocket is restricted, the possible variation in the position of the thrust axis is small. Therefore, in order to keep the main retro-rocket thrust alignment within allowable limits, the spacecraft lateral center of gravity must be controlled to within approximately 0.5 in. from the spacecraft geometric center line. With no payload, the lateral center of gravity of the basic bus is approximately 4.5 in. from the spacecraft center line. The payload selected (see Table 6) had to be so distributed that the spacecraft center of gravity would be repositioned to within the required tolerance. To achieve this objective on the reduced weight spacecraft, Compartment C (used to house scientific payload auxiliaries on the 2500 lb configuration) was eliminated and a special Compartment D added in the position normally occupied by the soil analysis instruments. (Compartment D is struc-

Table 6. Payload of reduced-weight spacecraft

Item	Weight, lb
Engineering instrumentation	10.0
Tape recorder and auxiliaries	10.6
Battery, special	5.0
Compartment D	4.0
Harness	4.0
Television (two cameras with color)	30.6
Television auxiliaries	1.3
Subsurface sampler and auxiliaries	39.5
Subsurface geophysical probe and auxiliaries	4.5
Surface temperature (interferometer spectrometer) and auxiliaries	2.2
Scientific thermal auxiliaries	5.3
Scientific mechanisms auxiliaries	2.0
Anchors	10.9
Total payload	129.9

turally and thermally the equivalent of Compartment C and does not constitute a new development.) This compartment will house the tape recorder and its auxiliaries, subsurface sampler auxiliaries, the subsurface geophysical probe auxiliaries, scientific mechanisms auxiliaries, and a special battery described below. None of these items is required to operate during the lunar night.

The remainder of the payload is installed in a manner identical to the installations presently being designed for the 2500-lb spacecraft. With the payload so distributed, the total spacecraft center of gravity falls within the required tolerances.

The electrical power and mission sequencing requirements during transit and lunar operations were investigated to ascertain whether or not the spacecraft can perform its mission with only one 45-lb battery carried on the basic bus. It was concluded that the only time power demands will exceed the capability of the 45-lb

battery is during terminal descent when the retro-engine ignition squibs and the retro-engine ejection squibs will be fired. The squib pulses are superimposed on a large steady-state load created by the flight control and terminal descent radar. It was concluded that the problem can be eliminated by using an auxiliary 5-lb battery to fire the squibs and thus a corresponding weight allocation has been added to the payload.

The engineering instrumentation listed in the payload summary represents instrumentation in addition to that previously planned for the 2500-lb spacecraft and is intended to assess in some detail the performance of the basic spacecraft system. The tape recorder will be used for engineering purposes during periods of the mission when data cannot be transmitted real time to the DSIF (i.e., boost, injection, touchdown).

The total spacecraft weight is shown in Table 7. Usable propellant weight of 1337.7 lb represents the combined vernier engine and retro-engine propellant weight. The total weight of the spacecraft as shown is 2099.3 lb.

Table 7. Summary weight statement

Basic bus	631.7
Usable propellant	1337.7
Payload	129.9
Total weight	2099.3 lb

This lighter configuration does require some minor design changes. However, because it is a simpler and less complicated device, by virtue of having fewer components, it is possible to adjust the time allocation among subsystem development spacecraft assembly, and systems test so that there is a high confidence in the spacecraft's performance and in meeting the specified launch dates.

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JPL SPACE PROGRAMS SUMMARY NO. 37-14

B. Scientific Payload

1. Introduction

Effort during this reporting period has been devoted primarily to problems associated with List I instruments defined in *Surveyor* Spacecraft System Design Specification 30240A as follows:

- (1) Television.
- (2) X-ray diffractometer.
- (3) X-ray spectrograph.
- (4) Gas chromatograph.
- (5) Geophysical surface subsystem.
- (6) Geophysical subsurface probe.
- (7) Magnetometer.
- (8) Plasma probe.
- (9) Seismometer.
- (10) Atmosphere pressure gauge.
- (11) Radiation detector.
- (12) Surface sampler.
- (13) Subsurface sampler.

Although work has continued in all areas to assure the over-all compatibility of the "basic spacecraft" and payloads consisting of List I experiments, particular emphasis has been placed upon the List II payload, which JPL has selected for Launches P-42 and P-43. This payload consists of the following experiments which were selected from List I.

- (1) Television.
- (2) X-ray diffractometer.
- (3) X-ray spectrograph.

- (4) Gas chromatograph.
- (5) Geophysical surface subsystem.
- (6) Geophysical subsurface probe.
- (7) Radiation detector.
- (8) Surface sampler.
- (9) Subsurface sampler.

During this period minor modifications to this list were made. The resistivity and density background counter portions of the geophysical surface subsystem were deleted from List II, and the Geiger counter portion of the radiation detector was added to List II.

The location chosen for the instrument portion of each of the above experiments is depicted in the general arrangement drawing, Figure 1 in Section A. The location of the electrical and mechanical auxiliaries can best be understood by reference to Figures 136, 137, and 139 of Section J.

A limited amount of effort has been devoted to the consideration of new experiments. In particular, JPL requested investigation of the problems associated with the integration of an alpha particle scattering instrument and a petrographic microscope into the basic spacecraft. Portions of each task have been completed and work is continuing in both areas.

2. Instrument Development

The third and final design reviews of most List I instruments currently being procured under JPL contract have

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been completed. Hughes Aircraft Company has continued technical assistance to JPL on all instrument contracts through this reporting period, and in general all instruments have reached the prototype stage of development. The one possible exception to this generalization is the radiation detector. Hughes Aircraft Company is currently negotiating a contract on the ionization chamber for this detector, and JPL has recently awarded a contract on the Geiger counter to Electro Optical Systems of Pasadena, California. It appears that all subcontracted List II instruments, with the above exceptions, have reached, or nearly reached, a stage of development such that the formal type approval testing can be undertaken with a high degree of confidence.

Progress on List II instruments under development at Hughes Aircraft Company (e.g., surface sampler, subsurface sampler, and TV) has been in consonance with other instruments requiring delivery to T-4, SC-1, SC-2, and SC-3. Details of the activities in those areas for this period may be found in Sections E and J of this report.

3. Systems Integration

For the past 2 mo, this activity has been devoted largely to the solidification of List II instrument auxiliary design and of all interfaces associated with the integration of List II instruments into the basic spacecraft. Of particular interest has been the collection of interconnection data to provide the basis for T-4 and SC-1, SC-2, and SC-3 electrical harness design, and the release of outline and mounting information to provide the basis for firm spacecraft general arrangement, mounting hardware and mechanisms design.

Circuit releases for nearly all List II electronic auxiliaries have been made during this period. Specific exceptions were interferometer spectrometer auxiliaries and Geiger counter auxiliaries. Subsurface magnetic susceptibility coils have been received from Texaco Experiment Incorporated. These coils and the surface magnetic susceptibility coils will be mated with their respective electronic auxiliaries, and the results will be reported during the next period. More detail on spacecraft electronics may be found in Section E of this report.

Development of the instrument mechanical auxiliaries has progressed with each instrument. This has in part been effected by the representation of mechanism design personnel at JPL vendor design reviews. Of particular interest has been the addition of certain verification signals, confirming the proper deployment of mechanisms and thus enhancing the confidence that individual scientific instruments have been properly deployed.

Recent investigations have revealed a potential mechanical integration problem with the seismograph. The problem is in part related to the maintenance of the proper payload center of gravity while at the same time assuring a successful deployment of the instrument. Modifications to the basic spacecraft and/or the instrument are being considered and results will be reported during the next period.

Thermal integration of the payload and the basic spacecraft has continued in all areas. In particular, a seismograph thermal control (*Sombrero*) study has been completed, and surface treatment of all List II instruments has been specified. Recommended surface treatment for some units requires the use of coatings developed at Hughes Aircraft Company during this period and which have absorptivity-to-emittance ratios in the order of 0.25.

Instrument thermal models required for MT-1 testing have been received from the vendors and are being prepared for test installation.

In addition to List I or II activities, work has been carried on in this report period to define the integration requirements of two List III units, an alpha particle scattering instrument and a petrographic microscope. These investigations will result in the determination of the electrical, mechanical, and thermal auxiliary functions that are required to allow each of these instruments to meet their over-all requirements for all phases of the scientific mission. Work in both areas is continuing.

4. Television

The television system provides visual surveillance of the lunar surface from the spacecraft. Three cameras (No. 1, 2, and 3) are used to survey the surface. Variable focal length lenses on the cameras provide wide or narrow angle capability. Scanning is accomplished by means of moving mirror assemblies on the cameras. One of the cameras (No. 2) also includes a filter wheel for color, polarized, or any other type of filtered pictures. A fourth camera, with fixed focal length lens, is located in a downward-looking position to obtain approach pictures of the Moon during the decent phase. For high-resolution pictures of certain selected areas, the spacecraft includes a fifth camera with a telescopic lens.

Each camera contains the electronic circuitry necessary to process the image formed on a vidicon tube. Two modes of operation, normal and emergency, are provided in each camera except the downward-looking approach camera, which has normal mode only. Under normal operation, single-frame pictures can be produced at a

maximum rate of one every 3.6 sec. The sweep circuits in the camera free-run continuously. Shutter operation and picture readout, including identification code, are initiated by command from Earth at the proper time in the sweep cycle and are synchronized by video logic circuitry in each camera. Prior to another shutter and readout cycle, time is allotted for vidicon erasure, effected by the continuous sweeping of the tube, and for operation of mirror and lens mechanisms by Earth commands. Control circuitry for the mechanisms also is included in each camera.

Emergency mode operation is similar to normal mode except for a reduction in sweep rate and open-shutter operation. In this mode, the picture rate is reduced to one in about 62 sec. The emergency mode will be used when a failure in the communications link severely reduces the available transmission bandwidth.

a. Progress summary. At this point in the spacecraft television development program, 95% of the electronic circuitry has been developed and released for product engineering. This circuitry accounts for over 200 transistors. The remaining circuitry lies in the area of thermal control, temperature sensing, and mirror control.

Breadboards have been made and tested for all circuitry released. Duplicate breadboards are now being integrated into a complete camera system. As prototype circuit modules become available, they will replace these system breadboards, thereby creating a more realistic, compact, prototype camera.

b. Vidicon procurement. As a result of evaluation studies early in the program, a decision was made to use an electrostatically focused, magnetically deflected vidicon because of its many advantages in this application. Subsequent to this basic decision, the principal vidicon procurement effort has been directed toward obtaining a suitable photoconductor surface for the vidicon. The general requirements for such a surface are as follows:

- (1) The surface must have adequate storage of a shuttered image to provide proper operation in both normal mode (1-sec frame readout) and emergency mode (20-sec frame readout).
- (2) After readout, the image must erase in the normal picture sequence to such a point that at re-exposure no noticeable trace of the previous image exists.
- (3) Sensitivity must be high to allow large lens f-numbers to be used in the presence of low light levels.
- (4) The surface should exhibit good dynamic range and be capable of detecting at least eight gray scales.

- (5) The surface must be capable of 72 hr sterilization soak at $+125^{\circ}\text{C}$.

Examination of these requirements reveals that no single commercial surface presently available can simultaneously meet all requirements. Consequently, it is apparent that either a new photoconductive material must be developed, or special circuit techniques must be employed to enhance performance. Hughes Aircraft Company presently is investigating both alternatives.

Three vendors, Westinghouse, GEC, and RCA, have been contacted in regard to their capability to produce new and suitable photosurfaces which might meet all requirements without necessitating the use of special erase techniques. Of the surfaces which have been examined, the most promising to date is the GEC antimony sulphide oxysulphide. This photosurface, proprietary with GEC, provides adequate storage, high sensitivity, good dynamic range, and excellent high-temperature capability, but is slightly deficient in that it requires five normal mode frames to erase adequately rather than three.

Special erase techniques also are under study. Washing the faceplate with light, greatly increasing the beam current during erase, and combinations of these two techniques are being examined carefully. Varying degrees of success have been achieved with various surfaces. However, improvements in erasure have usually been accompanied by a reduction in sensitivity and dynamic range.

c. Vidicon thermal control. Since the last report, extensive thermal studies have been made for the television cameras. These analyses revealed a potential thermal control problem if Peltier devices were used for vidicon faceplate temperature control; consequently, the vidicon thermal control technique has been changed. The new concept utilizes a special passive radiator for cooling the faceplate and, during low temperature ambient conditions, adds heat by means of a heater element.

The new vidicon thermal control block diagram is illustrated in Figure 4. The timing sequence of the logic circuitry is shown in Figure 5.

The initial warmup of the vidicon faceplate, prior to camera "On," begins when the unregulated $+22\text{ v}$ is gated to the control circuit by a power switch which closes upon receipt of the "Start Vidicon Thermal Control" command.

The basic thermal control circuit consists of a voltage regulator, temperature sensor, differential amplifier, level detector, power switch, and heater. During the initial

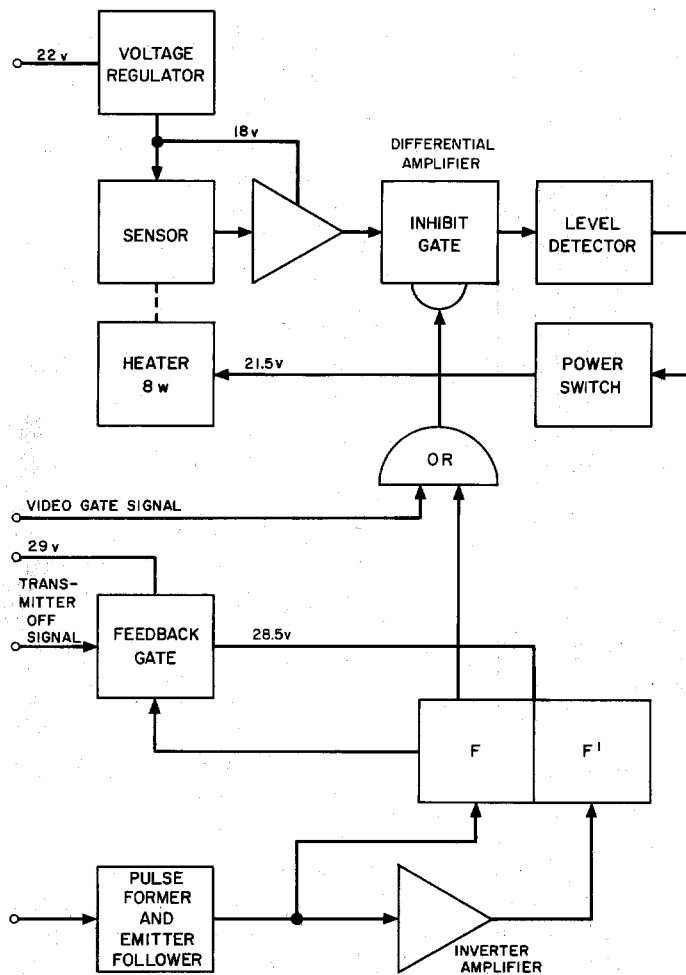


Figure 4. Vidicon thermal control

warmup period, these units form an uninhibited thermal control loop. The voltage regulator provides a stable voltage of +18 v for the sensor, differential amplifier, and level detector circuits.

Temperature sensing is accomplished by means of a passivated planar diode, which has a repeatable and consistent forward voltage versus temperature characteristic of approximately $-1.8 \text{ mV}/^\circ\text{C}$. The present design concepts are such that the sensor and heater will be mounted on a collar which will be placed on the vidicon tube neck approximately $\frac{1}{4}$ in. behind the collector ring.

The forward voltage of the sensing diode is compared to a reference voltage by the differential amplifier. A drop in sensor temperature causes the diode voltage to rise. This voltage rise is amplified by the differential amplifier, and the resultant output triggers the level detector. The level detector turns on the power switch, which gates unregulated 22-v power to the heater. Heat-

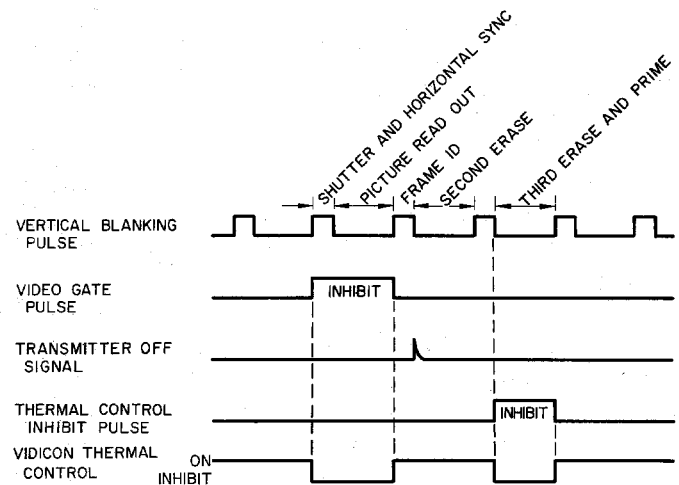


Figure 5. Vidicon thermal control logic

ing the sensor causes the reverse operation, and thus causes the power switch to be turned off. When in the "off" condition, the power switch draws no power. Because of possible anomalies introduced by the control system during camera operation, heating power is applied only during the 1.4-sec period following picture readout, as shown in Figure 5. Therefore, during a picture sequence, the thermal control circuit is inhibited by the video gate pulse during vidicon exposure and picture readout. It is also inhibited during the third erase and priming period by an inhibit pulse generated in the thermal control logic circuit. If the camera is on but not in a picture sequence, the thermal control loop operates uninhibited.

The logic circuit receives +29 v when the camera is turned on. The pulse former and emitter follower circuit is the only one that is immediately energized, which is necessary to prevent excessive loading of the vertical blanking pulse. Upon receipt of the "transmitter off" signal from the video logic circuit, the feedback gate turns on and applies power to the bistable multivibrator and inverter amplifier. The bistable multivibrator always turns on in the stable state such that no output appears at the OR gate. Upon receipt of the next vertical blanking pulse, the negative spike formed by differentiating the trailing edge of the vertical blanking pulse is amplified and inverted by the inverter amplifier, and is used to trigger the bistable multivibrator to its other stable state. The resulting output voltage is applied through the OR gate to the inhibit gate which then inhibits the application of power to the vidicon faceplate heater. Upon receipt of the next vertical blanking pulse, the leading edge is dif-

ferentiated and applied directly to the bistable multivibrator to trigger it back to its original state. This removes the voltage to the inhibit gate and turns off the feedback gate, thus returning the entire logic circuit to its original condition.

d. Iris control and automatic iris servo. The iris control system block diagram is shown in Figure 6. The purpose of this system is to control the light intensity on the faceplate of the vidicon. This is accomplished in the automatic mode by means of a closed loop servo, and in the manual mode by command. Since the manual mode positioning circuitry is identical to the other optics control circuits, described in a previous report, only the automatic servo is discussed here.

The automatic mode control is a form of dynamic positioning servo. A photo-cell (Clairex 605-IL photoconductive cell) is used as an optical radiation transducer to sense the total amount of light entering the lens system. A beam splitter provides a nominal 6% sample of the total light through the lens and focusses this sample on the sensitized surface of the photodiode.

The output of the photodiode is fed through an emitter follower to a differential amplifier for amplification and phase reversal. In the automatic mode, the two outputs of the differential amplifier are fed to two level detectors. These level detectors, one for clockwise and one for counterclockwise rotation, are set to turn on the iris adjust circuitry if the light input is too high or too low. A dead zone exists between the detector turn-on levels; this is necessary for stability because of the digital nature of the motor control. When the total light intensity changes more than 7% of the nominal value, the output of the differential amplifier triggers the appropriate level detector. The level detector activates the power switch which energizes the motor control circuit, causing the motor to drive the iris in the proper direction. When the light intensity returns to the proper level, the power switch de-energizes the motor control circuit.

There are two end-stop microswitches on the iris assembly which prevent overdriving of the iris when in the automatic mode of operation. Whenever a change in light level is greater than the dynamic range of the iris servo (too much or too little light of the iris to accommodate), Switch f/4 or f/22 will close. Closure of either limit switch causes the mode selection multivibrator to revert to the manual mode of operation, thereby automatically turning off the motor. The circuit has a third level detector, which works in manual mode to detect the presence of excess light which could damage the vidicon. The out-

puts of the three level detectors (two for the automatic mode, one for the manual mode) are connected to the shutter control logic to inhibit the shutter trigger signal when the light intensity exceeds preset levels. An override feature allows triggering of the shutter when the iris is at f/22 regardless of light intensity.

5. Tape Recorder

Development and manufacture of the tape recorder is under subcontract to the Data Recorders Division of Consolidated Electrodynamics Corporation.

The tape recorder is to be capable of recording and reproducing data, on command, during two operational phases of the spacecraft. During the boost and landing phases, engineering data will be recorded on seven analog channels (5 to 1000 cps); record and reproduce will be at 20 ips. For lunar surface operation, up to 18 hr recording time is provided at $\frac{1}{2}$ ips. Playback is at 20 ips. Three analog channels covering the range $\frac{1}{60}$ to 5 cps, and one analog channel for the range $\frac{1}{20}$ to 20 cps, are provided for recording seismic activity. Two digital channels (NRZ) are provided for recording time-shared data from the plasma probe and seismometer (tide data) and for the magnetometer.

The very low-frequency analog response (down to $\frac{1}{60}$ cps) will be achieved through the use of saturated FM recording techniques. Input voltages will drive voltage-controlled oscillators with center frequencies of 333 cps or 20 kc (lunar or launch) with deviation of $\pm 50\%$. Driver amplifiers will amplify the frequency modulated signals and saturate the magnetic tape. For the digital channels, solid-state switches will feed input to the driver amplifiers for saturation recording of the digital signals.

In the response mode, signals from the playback head will be linearly amplified by high-gain amplifiers and fed through level detectors (Schmitt Trigger), which will shape the signals. Each pulse of the frequency-modulated signals then triggers one-shot multivibrators which are designed to have constant area pulse regardless of frequency. Active filters then integrate the signals to obtain their average values, which are proportional to the original input signals. The signals are then amplified to the required output level. The same reproduce amplifiers are used for both lunar and launch modes. Lunar digital signals are obtained directly from the output of the level detectors.

The capstan motor servoamplifier circuit, and the control circuitry which provides electronic mode switching in response to commands from Earth, are being revised and will be described in a future report.

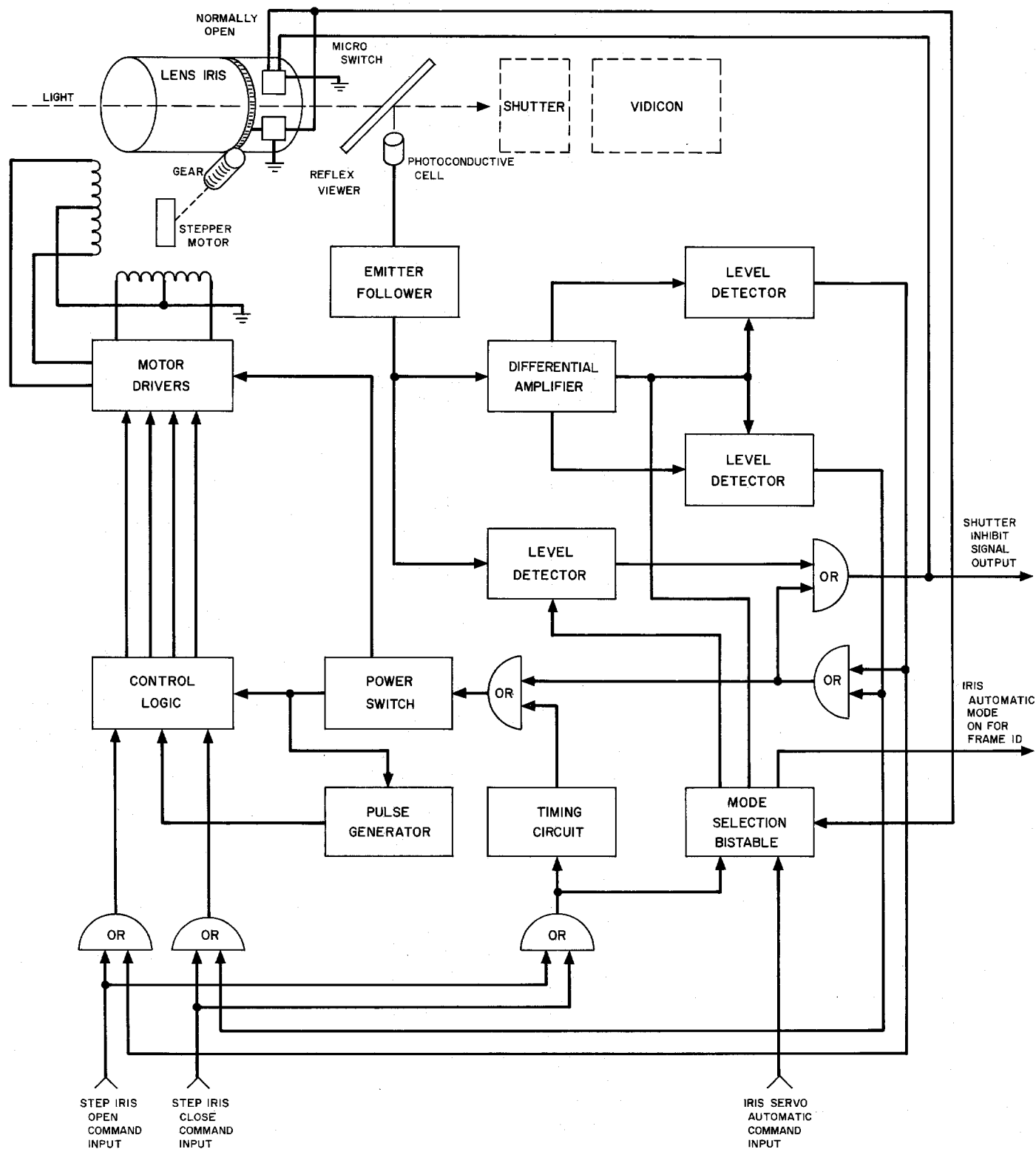


Figure 6. Iris control system

All circuits necessary for construction of the breadboard have been developed; however, circuit development is continuing to incorporate temperature compensation, to optimize the circuits, particularly by reduction in number of components, and to reduce power requirements.

The electronics (Fig 7) and electromechanical breadboards were delivered in mid-February. Electronics are constructed on standard printed connector cards mounted in a 20-card rack.

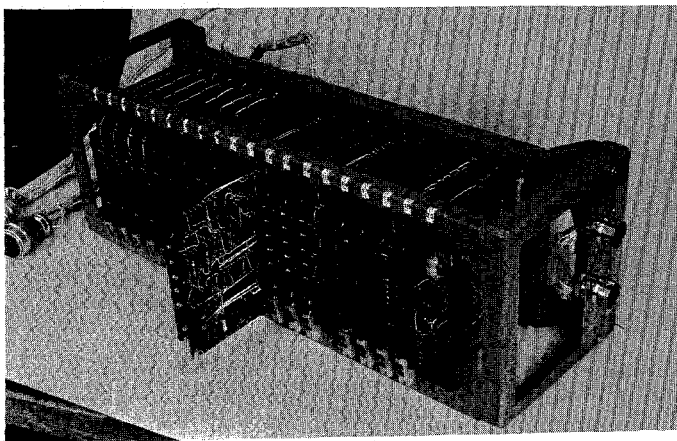


Figure 7. Tape recorder electronics breadboard

Electromechanical development has been principally in design, construction, and testing of the breadboard tape transport including reels, heads, differential, magnetic clutch, transmission, and motor. Difficulties experienced with initial models of special magnetic clutch and reel assembly differentials obtained from subcontractors have been corrected. Excessive noise was contributed by full range spider gear bearings in the differentials; this difficulty has been corrected by the use of radial-retainer bearings.

The electronic and electromechanical breadboards are being tested together as a complete system. Seven-channel recording in the lunar mode is satisfactorily being done with 700 mw input power. By decreasing voltage to the motor and through circuit refinements, particularly in the servo control circuit, the power is expected to drop below the 650 mw allowed.

Electronic and electromechanical developments are proceeding satisfactorily and the operating breadboard will be delivered in mid-February. Emphasis will now be on system testing, finalizing circuits for release to packaging, environmental testing of critical subassemblies, product design, and reliability studies.

The general packaging arrangement of the tape recorder has been determined. The cast base and drawn cover are of aluminum, to be soldered for hermetic seal. Designs of all mechanical parts are being reviewed for possible weight reduction.

6. Mechanisms for Scientific Payload

The mechanisms used in conjunction with the scientific payload are discussed in this section; mechanisms employed on the basic spacecraft bus are covered in Section J.

a. TV mirror assemblies. The TV mirror assemblies, providing azimuth and elevation scan for the TV cameras and for the high resolution telescope, are in developmental testing. A test setup has been made to obtain settling time data under ambient conditions (Fig 8); initial results indicate that the design provides good damping and consequently short-settling times after positioning in azimuth (Fig 9).

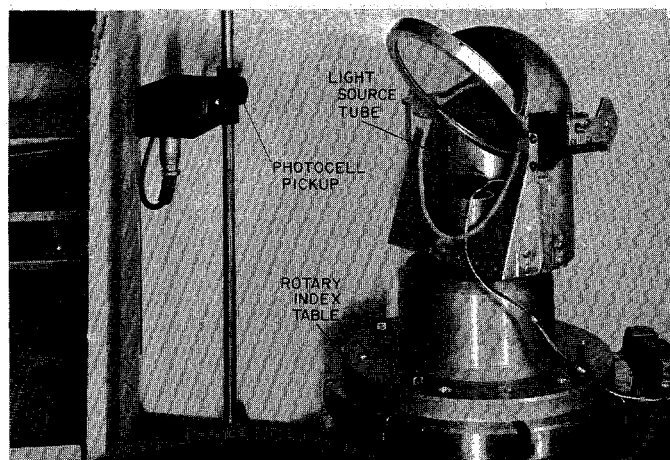


Figure 8. Mirror settling time testing

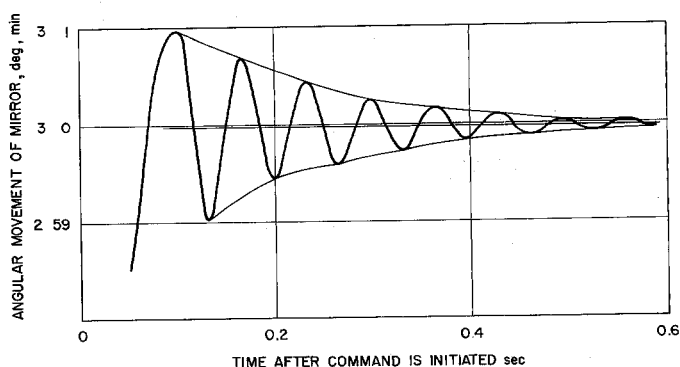
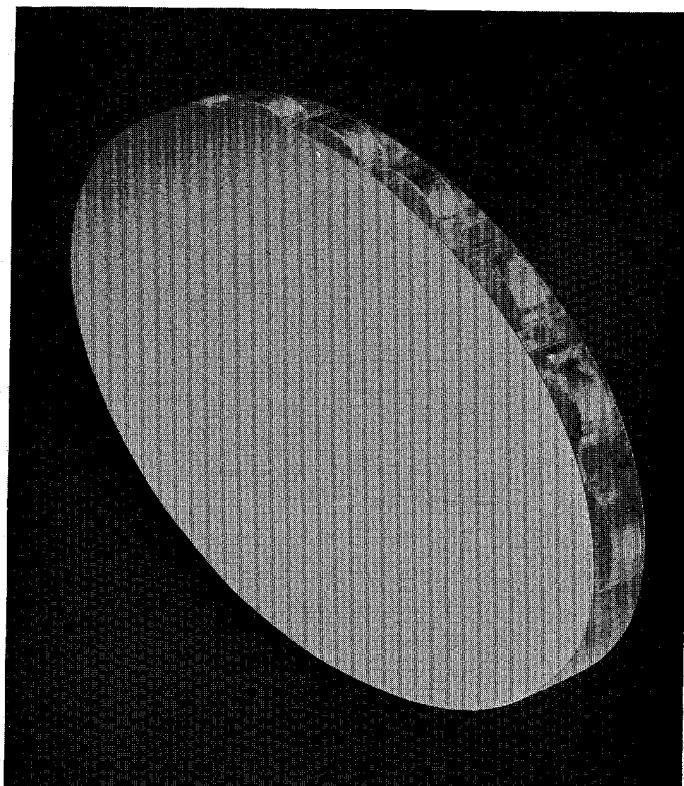
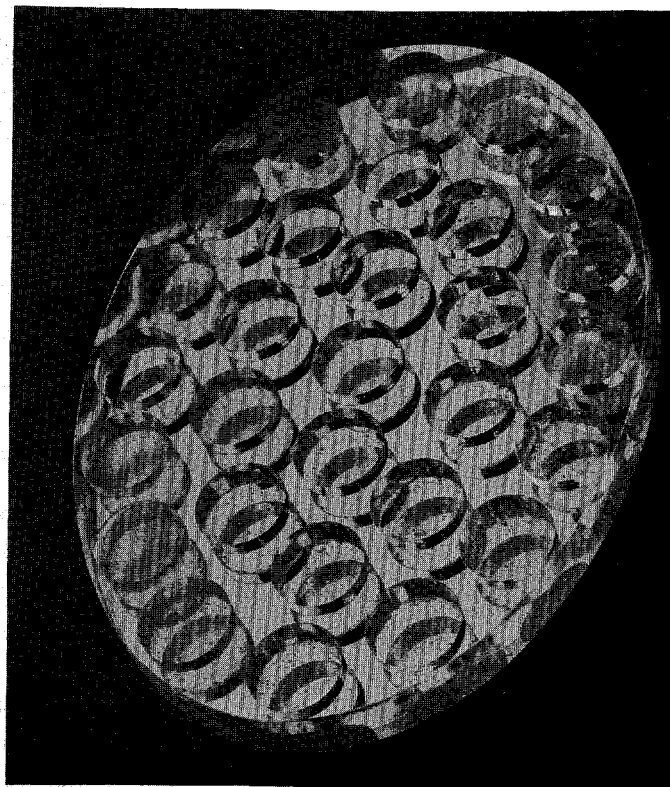


Figure 9. Mirror settling time for single azimuth step



a) Front



b) Back

Figure 10. Fuzed quartz mirror

The elevation worm gear drive assembly has been redesigned to provide easier assembly and more accurate adjustment of the worm and gear wheel mesh. The new design consists of a shim and dowel assembly which can be checked as a subassembly prior to coupling onto the motor shaft.

A fuzed quartz mirror having an aluminized front surface and silicon monoxide overcoating has been received for evaluation. This mirror is $\frac{1}{2}$ in. thick but is cored on the back side to reduce weight, resulting in a weight equal to that of a solid $\frac{3}{8}$ in. thick pyrex mirror (Fig 10).

A solid beryllium mirror has been ordered for evaluation and is scheduled for delivery in mid-February. Due to the high modulus of elasticity of this material, the mirror may be self-supporting, and the mounting provisions may be incorporated in the mirror structure, eliminating the need for separate support hardware. This in turn allows the use of a thicker, more thermally stable mirror without an increase in over-all assembly weight.

b. Instrument extension booms. Extension booms are used to deploy the radiation detector and surface density

experiments. Initial developmental tests have commenced on an eight-section telescoping boom. During these tests the boom was extended using a simulated cold gas supply of helium pressurized to 5000 lb/in.². Extension velocity, acceleration, and plenum pressure were monitored and recorded. Preliminary test results indicate that seals are required at the boom section interfaces to prevent excessive gas leakage and to permit the extension regulator valve to function properly. A complete boom assembly incorporating teflon seals and a redesigned regulator valve has been fabricated and is now ready for test.

c. Magnetic susceptibility positioner. The magnetic susceptibility experiment requires that the instrument be deployed from the stowed position on the spacecraft to a point at least 12 in. above the lunar surface and not closer than 18 in. to the metallic spaceframe for a calibration measurement. It is then required to be lowered to the lunar surface at a controlled velocity. The mechanism has been designed to perform this function. Operation is initiated by the activation of an explosive pin puller, unlocking the mechanism which is driven up and rotated away from its stowed position on the spacecraft leg by two springs. After the calibration measurement,

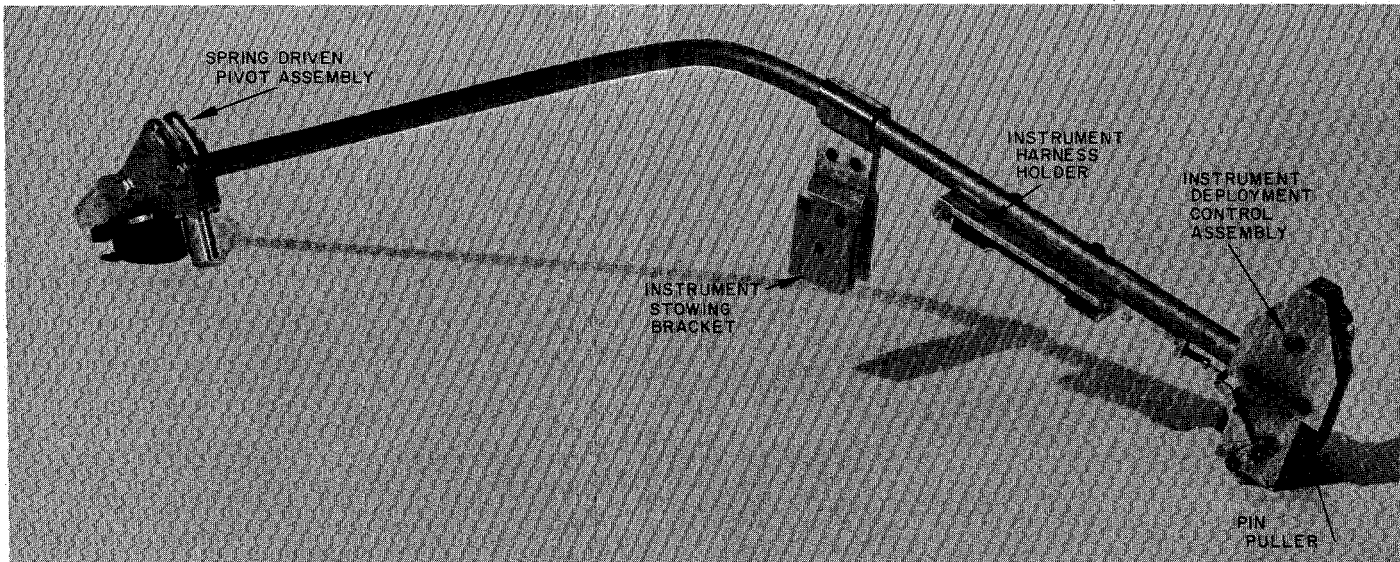


Figure 11. Magnetic susceptibility extension mechanism

another pin puller is actuated to allow the instrument to descend to the lunar surface. The descent is regulated by a friction controlled spool upon which teflon coated glass fiber cord is wound. The developmental model of this mechanism (Fig 11) has been fabricated and testing is scheduled to begin in the near future.

d. Thermal diffusivity positioning mechanism. This mechanism is required to transfer the thermal diffusivity experiment from the stowed position on the spacecraft to an operating position on the lunar surface. The instrument must then be rapidly rotated through an azimuth angle of approximately 150 deg.

A developmental model of the positioning mechanism has been fabricated (Fig 12). The mechanism operation is initiated by activation of two explosive pin pullers, allowing a coiled spring and gravity to pivot the device to a vertical position so that the instrument can then descend to the operating position above the lunar surface by the extension of two square telescoping tubes. Once in position, the instrument is pivoted in azimuth about a stand-off spike by a bidirectional stepping motor.

e. Acoustic source positioner. The mechanism designed for emplacing the acoustic source on the lunar surface consists of a titanium accordion-type spring isolated from both the spacecraft and the instrument by teflon couplings. Because the weight of the instrument is insufficient to assure that the wiring harness will be reliably deployed from its stowed position by gravity forces, the spring has been included to force the instrument to the lunar surface. The spring is compressed in the stowed

position (Fig 13, top) and is released by actuation of an explosive pin puller. The extension of the spring carries the instrument and its associated cabling to the local surface (Fig 13, bottom). A developmental model has been fabricated and is ready for environmental testing.

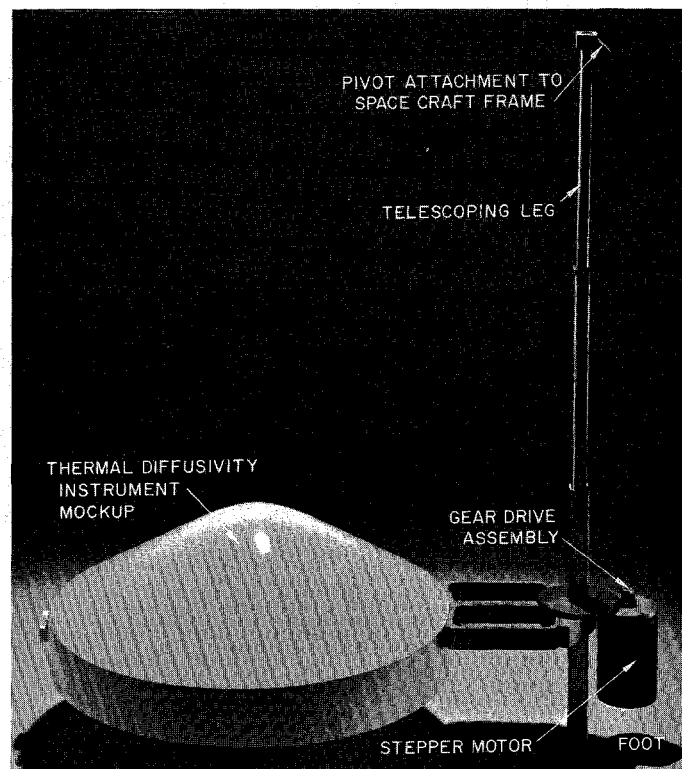


Figure 12. Thermal diffusivity positioning mechanism

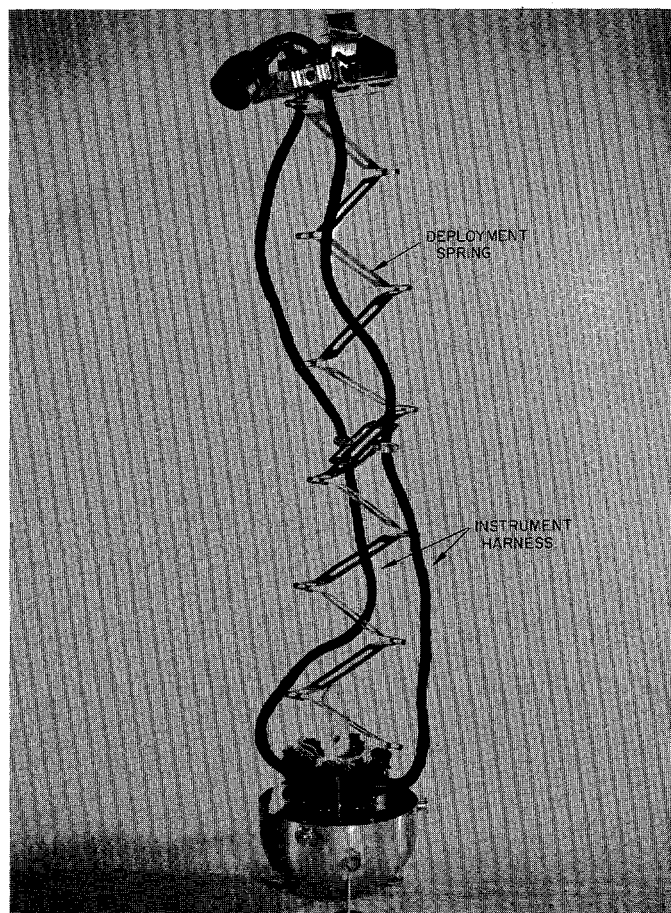
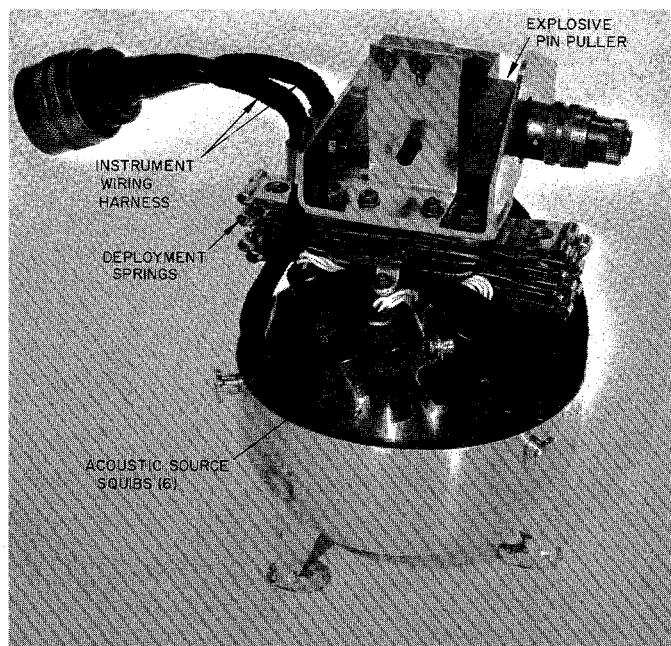


Figure 13. Acoustic source positioner in stowed (top) and deployed (bottom) position

f. *Acoustic sensor positioner.* The acoustic sensor positioner consists of a flat titanium arm pivoting on a spacecraft leg. The upper end of the arm is riveted to a teflon isolator which acts as a bearing for the rotation of the arm and instrument assembly. The lower end of the arm is also riveted to a teflon component which serves as the bearing for an explosive pin puller and provides an isolating attachment for the bracket that retains the acoustic sensor. Activation of the pin puller releases the swinging arm, allowing the instrument to fall to the lunar surface. A developmental model (Fig 14) of this device has been fabricated and is being introduced into developmental testing.

g. *Penetrometer extension mechanism.* The three penetrometer extension mechanisms are each mounted on the side of a spacecraft leg. The mechanism is designed to

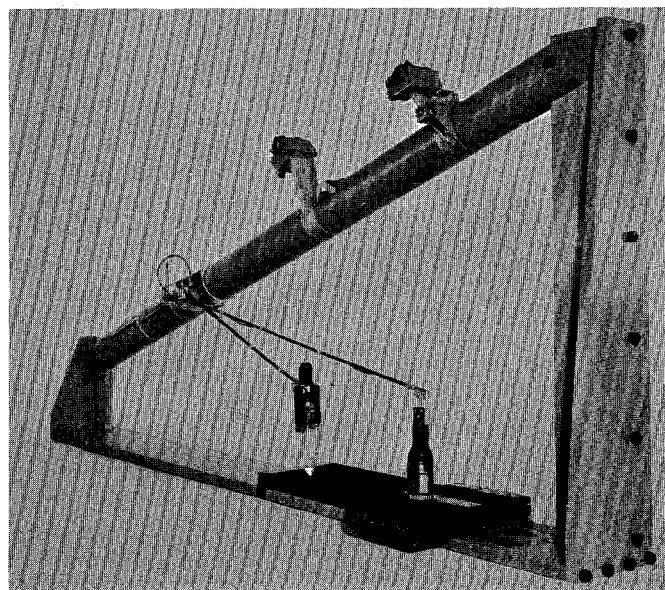
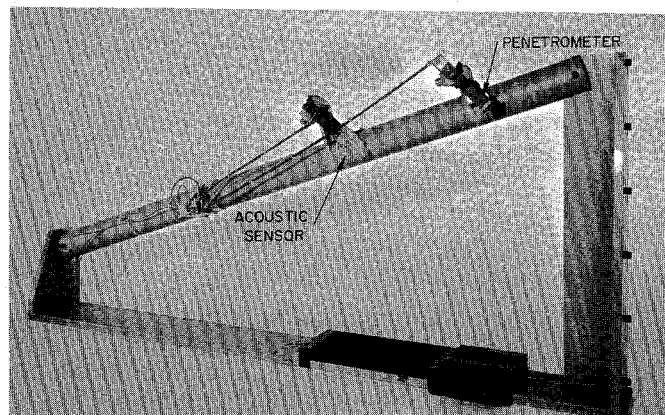


Figure 14. Acoustic sensor positioner and penetrometer extension mechanism in stowed (top) and operating (bottom) position

provide vertical tension for instrument deployment through a single flexure hinge. The mechanism consists of a flat titanium arm which is V-formed at its midsection and coupled to both the spacecraft and the instrument through teflon isolators. On one end of the arm, the teflon isolator provides a bushing support for an explosive pin puller and a coupling to the instrument bracket. On the other end of the arm, the flat spring section is riveted to the teflon isolator which in turn is fastened to the spacecraft. Operation is accomplished by actuation of the explosive pin puller, releasing the instrument from its stowed position and allowing the preloaded single flexure hinge to force the instrument to the lunar surface. Developmental units have been fabricated and preliminary developmental tests in laboratory environment have been performed. Figure 14 (top) shows the acoustic sensor positioner and penetrometer extension mechanism in the stowed position. Both devices are shown in the operating position in Figure 14 (bottom).

h. Subsurface sampler. The subsurface sampler (Fig 15) drills a hole in the lunar surface to a maximum depth of 5 ft, delivers samples from any selected depth to the sample processor, and lowers a geophysical probe into the hole for measurements at selected depths. The sampler comprises a drill head, drill shank, bit, dump mechanism, vertical drill drives, supporting structure, subsurface geophysical probe, lowering drive, and jettison device.

A developmental model is nearing completion. This model has been designed to withstand increased landing loads, but this redesign has been accomplished within the specified weight limits.

Drilling tests with the breadboard model have been continued. Previous rock drilling tests had reached a maximum depth of 8 in. in atmosphere and 6¼ in. in a vacuum and had demonstrated that the drilling rate was not significantly affected by vacuum operation. A 5-ft block of berea sandstone was sunk into the ground at the Culver City drill test site. Drilling progressed smoothly until a depth of 12 in. was reached. At this point the drilling rate fell off and drill shank binding began to be troublesome. Study of the drill shank indicated that all of the wear and serious friction were taking place in the first 3 in. of the shank and were caused by poor scavenging of the drill products from the hole. Minor modifications of the drill shank solved this problem, completely eliminated drill shank binding, and produced an immediate increase in drilling rate. A 60-in. hole was drilled in the sandstone block in 67 min of accumulated drilling time.

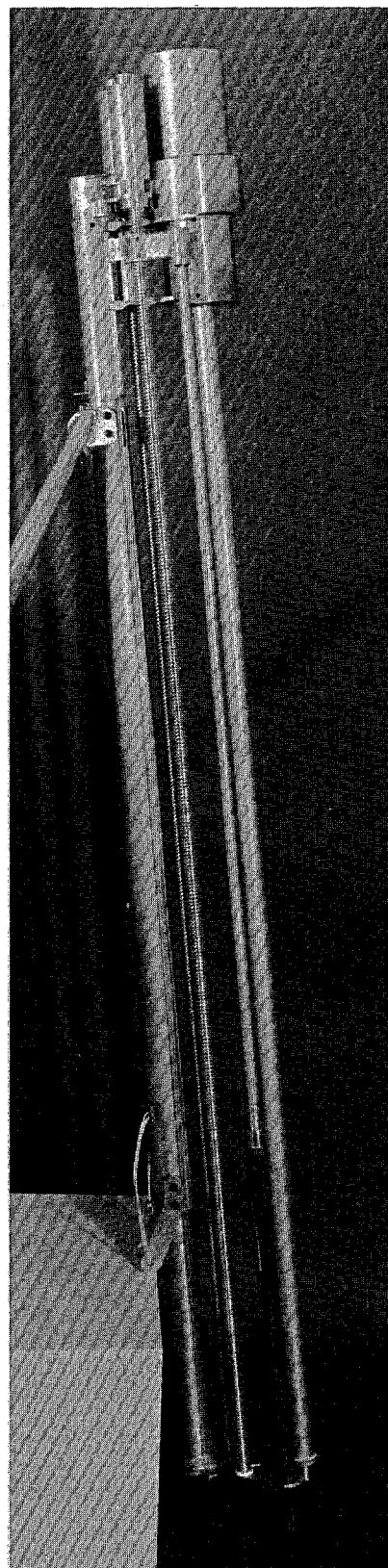


Figure 15. Subsurface sampler

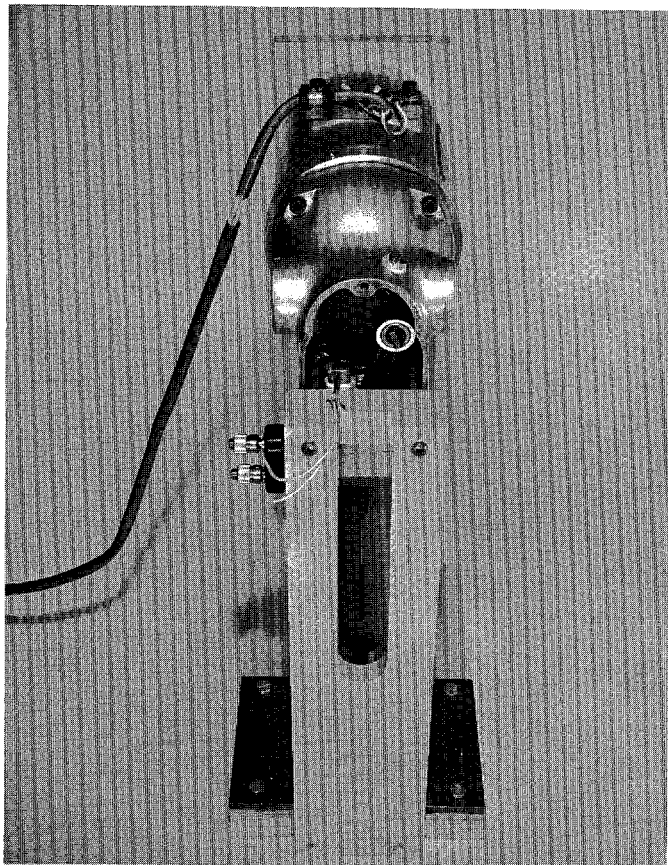


Figure 16. Drill head testing

The next drilling test to be performed is to drill a 21-in.-deep hole in Harris granite. The current drill rate in granite is approximately $\frac{1}{4}$ in./min. To meet all specification requirements, the drilling rate must be increased to $\frac{3}{8}$ in./min. A study is now being made of the drill operation with the objective of improving the drill rate in granite to at least $\frac{3}{8}$ in./min. Analysis of past drill test experiments with the Hughes drill bit indicate a possible 1 in./min drill rate in granite. Figure 16 shows an experimental drill head mounted so that stroboscopic and high-speed motion picture studies may be made of the hammer action on the drill shank anvil. Current measurements indicate that a 20 to 30% increase in drilling rate can be expected from a two-piece combination aluminum and titanium drill shank which will provide better matching of the hammer and drill shank masses. The modified drill shank has been designed and is being fabricated. Further studies are concerned with decoupling the central torque transmitting drill shank from the drill bit with a spring. This can be done because the hammer blow is entirely transmitted to the drill bit through the rigid outer drill tube. This is expected to reduce the apparent

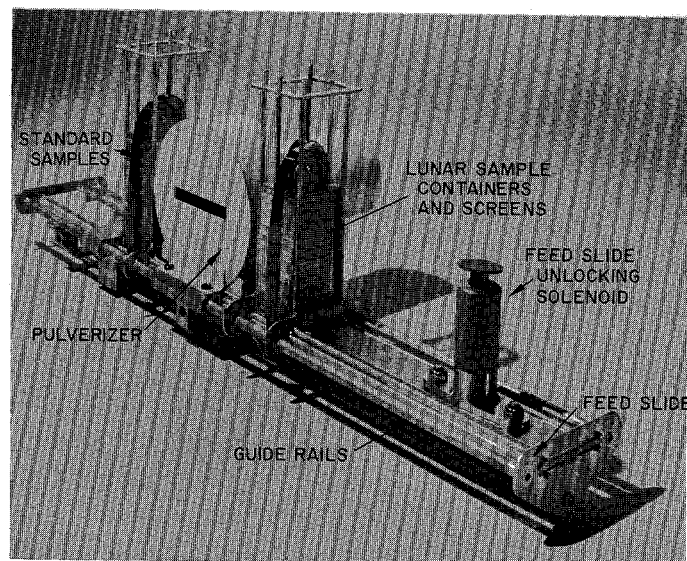


Figure 17. Sample transport system mockup

mass of the drill bit and shank assembly and improve energy transfer from the hammer.

The subsurface sampler system operational sequence data have been compiled. An electromechanical mockup of the system is now in construction for checking system operation control logic circuits and fail-safe provisions in the event command communications are interrupted.

An alternate mechanism design for the operation of the drive head, bit, and probe has been proposed by Arthur D. Little Co. This design, which employs a cable system to provide drill feed, dumping, and probe insertion, is currently being studied and compared with the present feed screw system.

i. Sample processing system. Samples of lunar soil received from the surface or subsurface samplers are accurately measured and delivered to the pulverizer or

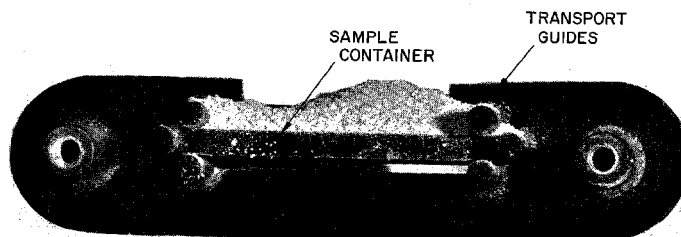


Figure 18. Sample container and transport guides

directly to the gas chromatograph by the measure and transfer device. A pulverizer reduces the sample to the particle size specified for X-ray instruments, and a sample container transport system screens the sample and delivers it to the X-ray spectrometer or X-ray diffractometer. A mockup sample transport system is shown in Figure 17.

The tubular construction of the transport guides is self-clearing, and rock powder or sand cannot accumulate and cause jamming. The $\frac{1}{8}$ in. and $\frac{1}{4}$ in. aluminum tubes are hard-anodized. Several thousand cycles of operation of the guides and sample containers covered with crushed quartz particles as shown in Figure 18 did not damage the hard anodizing or cause appreciable wear.

Figure 19 shows an unrelieved sample container, relieved sample container, and transport test fixture. Tests were performed with the guides covered with rock powder in atmosphere and in a 10^{-4} mm Hg vacuum (Fig 20). A driving force of 9 lb was required to move the unrelieved container along the rails and a maximum of only $1\frac{1}{4}$ lb was required to move the relieved container, resulting in increased reliability.

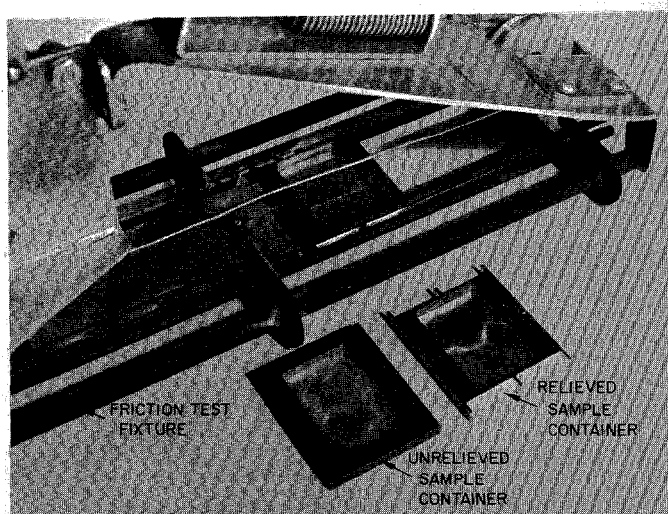


Figure 19. Sample transport test fixture

Friction tests of this system have demonstrated that the length of the slide-to-rail contact is the most critical factor affecting the force required for operation in the presence of rock powder. Any clearance between the slide and guide rail can present a tapered entrance to the rock powder. Crushing the first rock particle produces several more particles in the tapered gap. As the

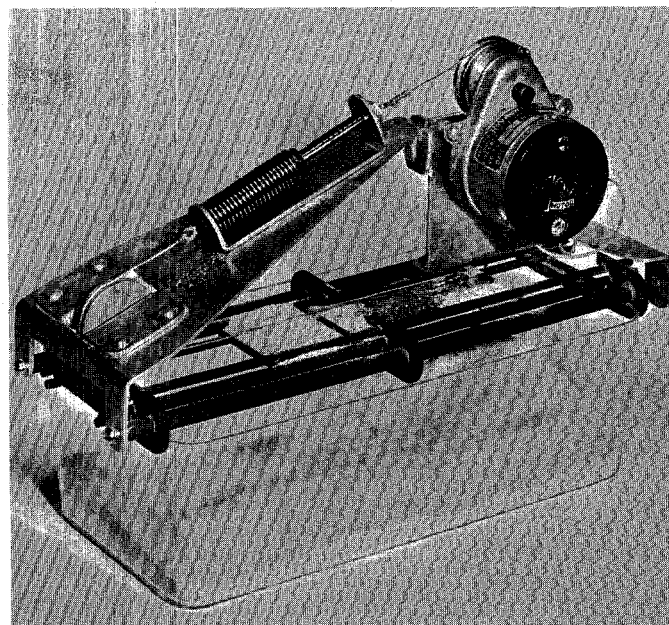


Figure 20. Testing of transport guides

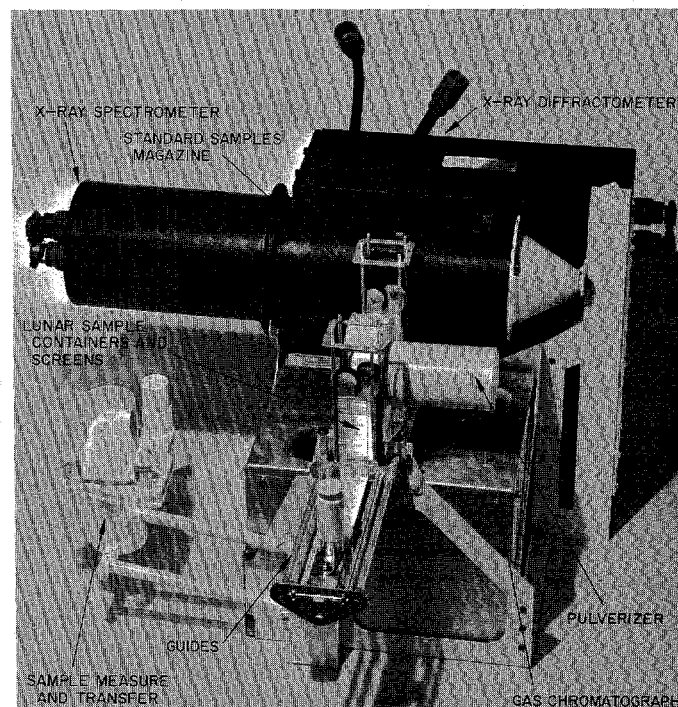


Figure 21. Soil analysis group

slide advances and the rock particles are crushed, the number of particles in the gap and the force required to crush them increases exponentially with slide-to-guide contact length.

A mockup of the lunar soil processor packaging and support structure is shown in Figure 21. The sample

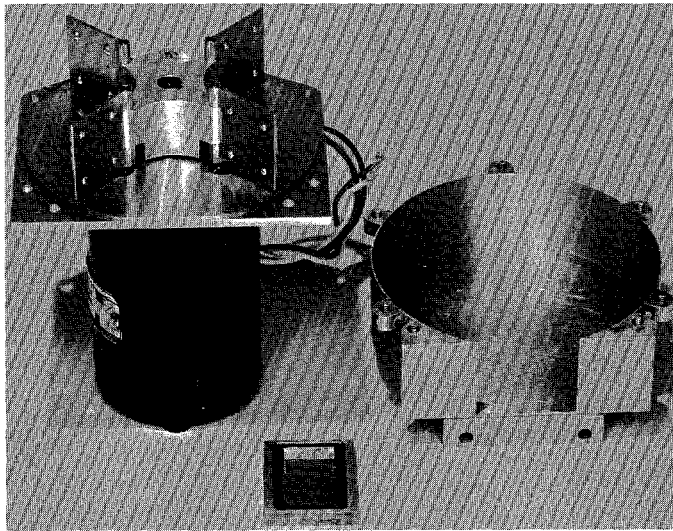


Figure 22. Development pulverizer

transport mechanism, pulverizer, gas chromatograph, and X-ray diffractometer are supported on this structure. Several other designs were mocked up in an attempt to support the instruments economically in an external framework adapted to the present instrument mounting provisions. An approach to this problem using the instrument cases as support structure will save 60% of the mounting structure weight. This structure will take 30 g

landing load and will weigh less than 1½ lb as compared to a 5-lb present weight. The increase in instrument case weight will be less than ½ lb. This design concept is being studied.

The pulverizer is a high-speed rotary particle impact device, and by actual laboratory tests will reduce 4 cc of Harris granite to —200 mesh in less than 45 sec. The developmental pulverizer (Fig 22) is designed to evaluate the biasing from preceding samples and the degree of wear contamination in the samples pulverized, screened, and compacted in the soil sample container.

Three types of pulverizer interior coatings are scheduled to be tested in a vacuum chamber:

- (1) Hard copper coating, 0.010 in. Copper plate is the coating presently being used because copper shows only a narrow band in the X-ray measurements.
- (2) Teflon coating, 0.010 in. Teflon has properties that may allow a very low bias and contamination rate. Testing will establish the efficiency of the pulverizing action with a soft liner. Teflon is transparent to the X-ray instruments.
- (3) Hard anodize, 0.008 in. The extreme hardness of this coating makes it attractive for this application. There is possibility of a very low percentage of contamination, possibly as low as 0.1%.

C. System Analysis

1. Method of Midcourse Correction for Maximizing Over-All Vernier Fuel Margin

The "critical plane" method has been used for estimating the midcourse correction fuel requirement, because it is the minimum correction of miss that is needed for a given set of injection errors to be corrected at a given time. Study of the combined midcourse and terminal descent system characteristics, however, reveals that the critical plane correction is not necessarily the optimum maneuver from the over-all fuel standpoint when the miss is large. In fact, the vernier system fuel margin may actually be increased by correcting all or part of the impact velocity error whenever the impact velocity tends to be too high. The following discussion outlines a fairly simple method of optimization which may be employed.

a. Midcourse fuel costs.

Critical plane correction. For the critical plane correction, the midcourse fuel cost is calculated through the following procedure. Given an impact parameter plane miss vector (M_1, M_2) , the required incremental velocity vector $\bar{V}_c$ is

$$\bar{V}_c = - \begin{vmatrix} k_{11} & k_{12} & k_{13} & -1 & M_1 \\ k_{21} & k_{22} & k_{23} & & M_2 \\ k_{12}k_{23} & k_{13}k_{21} & k_{11}k_{22} & & 0 \\ -k_{13}k_{22} & -k_{11}k_{23} & -k_{12}k_{21} & & \end{vmatrix} \quad (1)$$

where

$$k_{ij} = \frac{\partial M_i}{\partial \dot{x}_j} \quad i = 1, 2; \quad j = 1, 2, 3 \quad (2)$$

is the partial derivative of the i^{th} miss coordinate due to a velocity increment in the x_j direction at the time of the midcourse correction. Thus, the magnitude of $\bar{V}_c$ will be of the form (assuming x_j to be Cartesian)

$$V_c = (\dot{x}_{c1}^2 + \dot{x}_{c2}^2 + \dot{x}_{c3}^2)^{1/2} = V_c(M_1, M_2) \quad (3)$$

In practice, such a maneuver also changes the impact velocity slightly from the uncorrected case. The *amount of lowering* of V_{impact} is designated $(\Delta V_{\text{impact}})_c$.

Generalized V_{impact} correction. The fuel requirement was examined for lowering V_{impact} from the uncorrected case by ΔV_{impact} , the value of which will be found later based on the criterion of maximizing the over-all system fuel margin.

$$\bar{V} = - \begin{vmatrix} k_{11} & k_{12} & k_{13} & -1 & M_1 \\ k_{21} & k_{22} & k_{23} & & M_2 \\ k_{31} & k_{32} & k_{33} & & \Delta V_{\text{impact}} \end{vmatrix} \quad (4)$$

where

$$k_{sj} = \frac{\partial}{\partial \dot{x}_j} (V_{\text{impact}}) \quad (5)$$

The magnitude of V is

$$V = (\dot{x}_1^2 + \dot{x}_2^2 + \dot{x}_3^2)^{1/2} = V(M_1, M_2, \Delta V_{\text{impact}}) \quad (6)$$

Difference in midcourse fuel cost. The difference in fuel cost is

$$W - W_c = (V - V_c) \frac{\partial W}{\partial V} \quad (7)$$

$\frac{\partial W}{\partial V}$ = amount of midcourse fuel required per unit change in midcourse velocity increment

b. Effect on terminal descent fuel requirement. The criterion for judging the effect on the terminal descent fuel requirement is the main engine burnout velocity. For the critical plane correction, the burnout velocity is *lowered* by an amount essentially equal to

$$(\Delta V_{BO})_c = (\Delta V_{\text{impact}})_c + \left(V_c \frac{\partial W}{\partial V} \right) \frac{\partial V_{BO}}{\partial W_o} \quad (8)$$

where $\frac{\partial V_{BO}}{\partial W_o}$ is the sensitivity of the burnout velocity with respect to initial weight at the start of the main retro phase.

For the generalized V_{impact} correction,

$$(\Delta V_{BO}) = (\Delta V_{\text{impact}}) + \left(V \frac{\partial W}{\partial V} \right) \frac{\partial V_{BO}}{\partial W_o} \quad (9)$$

Thus, the difference in ΔV_{BO} is

$$(\Delta V_{BO}) - (\Delta V_{BO})_c = (\Delta V_{\text{impact}}) - (\Delta V_{\text{impact}})_c + (V - V_c) \left(\frac{\partial W}{\partial V} \right) \left(\frac{\partial V_{BO}}{\partial W_o} \right) \quad (10)$$

There is a coefficient $\frac{\partial W_f}{\partial V_{BO}}$ which may be evaluated for the vernier descent phase, which is the rate of change of the fuel required per unit change in the burnout velocity. Thus, the amount

$$\left[(\Delta V_{BO}) - (\Delta V_{BO})_c \right] \frac{\partial W_f}{\partial V_{BO}} \quad (11)$$

represents the net *decrease* in vernier phase fuel requirement for the generalized case over the critical plane case. However, this is achieved at a cost of increased midcourse fuel requirement equal to that given by Equation 7. The actual gain in fuel margin ΔW_{margin} is therefore

$$\Delta W_{\text{margin}} = \left[(\Delta V_{BO}) - (\Delta V_{BO})_c \right] \frac{\partial W_f}{\partial V_{BO}} - (V - V_c) \frac{\partial W}{\partial V} \quad (12)$$

$$= \left[(\Delta V_{\text{impact}}) - (\Delta V_{\text{impact}})_c \right] \frac{\partial W_f}{\partial V_{BO}} - (V - V_c) \frac{\partial W}{\partial V} \left[1 - \left(\frac{\partial V_{BO}}{\partial W_o} \right) \left(\frac{\partial W_f}{\partial V_{BO}} \right) \right]$$

If ΔW_{margin} is positive, the implication is that we are actually gaining in fuel margin by making a correction for V_{impact} that is larger than that achievable with the critical plane correction. As ΔV_{impact} increases, ΔW_{margin} will reach a maximum and then start to decrease pending on the value of

$$\left(\frac{\partial V_{BO}}{\partial W_o} \right) \left(\frac{\partial W_f}{\partial V_{BO}} \right)$$

being less than unity, which is true in the case of *Surveyor*.

The optimum correction (that which maximizes ΔW_{margin}) may be found by solving the equation

$$\frac{d(\Delta W_{\text{margin}})}{d(\Delta V_{\text{impact}})} = 0 \quad (13)$$

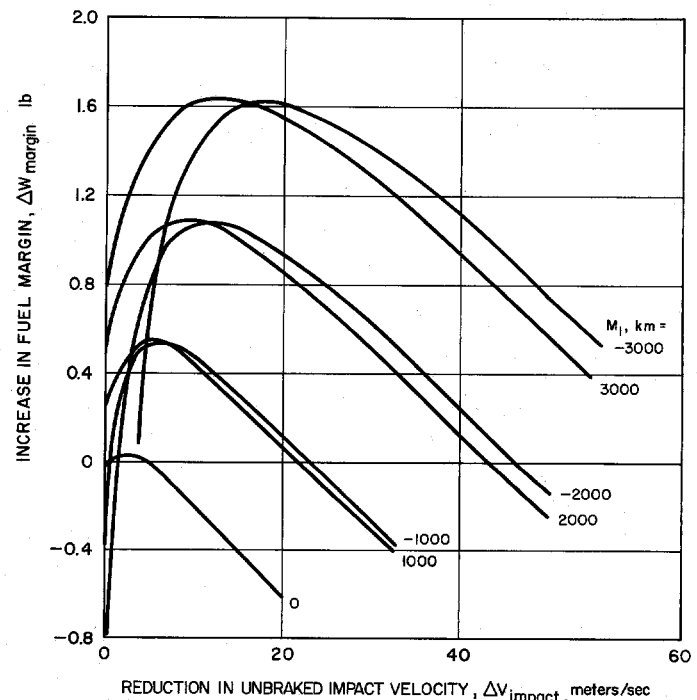


Figure 23. Increase in fuel margin vs reduction in unbraked impact velocity for 2000-lb spacecraft

for ΔV_{impact} and then substituting ΔV_{impact} into Equation 4 for $\bar{V}$. In general, this will involve the solution of a quadratic equation in ΔV_{impact} , the coefficients of which involve M_1 and M_2 . Thus, the optimum ΔV_{impact} will be dependent on the actual miss components.

c. Quantitative examples. Figure 23 is the plot of ΔW_{margin} against ΔV_{impact} (reduction) for various values of M_1 for a 2000-lb version of the *Surveyor* launched on a particular day. M_2 is set equal to zero. The curves all show a clearly indicated maximum for reasonable values of ΔV_{impact} . In Figure 24 similar conditions are plotted for a 2500-lb spacecraft. While the maxima are not shown, they do exist at very high and impractical values of ΔV_{impact} . This difference between the two sets of curves results from the fact that for the 2500-lb case, the product $(\partial V_{BO}/\partial W_o)(\partial W_f/\partial V_{BO})$ is very close to unity (about 0.92) whereas for the 2000-lb case, it is only about 0.6.

While the results indicate that the optimum correction from the fuel-saving point of view may not always be attained they do in general show that there is definite advantage in correcting impact velocities in excess of nominal. Further work is being carried out in forming a suitable midcourse logic to take advantage of this fact, which would take into account also the arrival time as well as terminal sensor constraints.

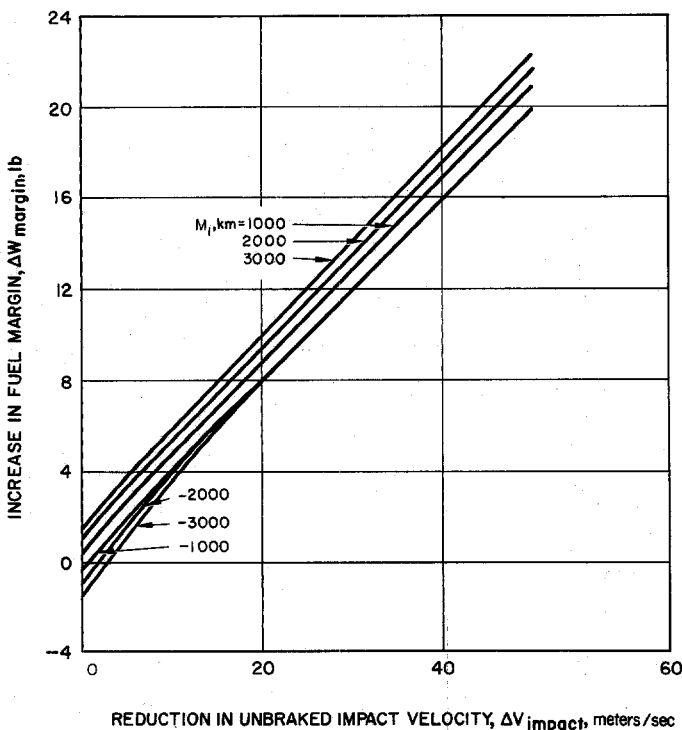


Figure 24. Increase in fuel margin vs reduction in unbraked impact velocity for 2500-lb spacecraft

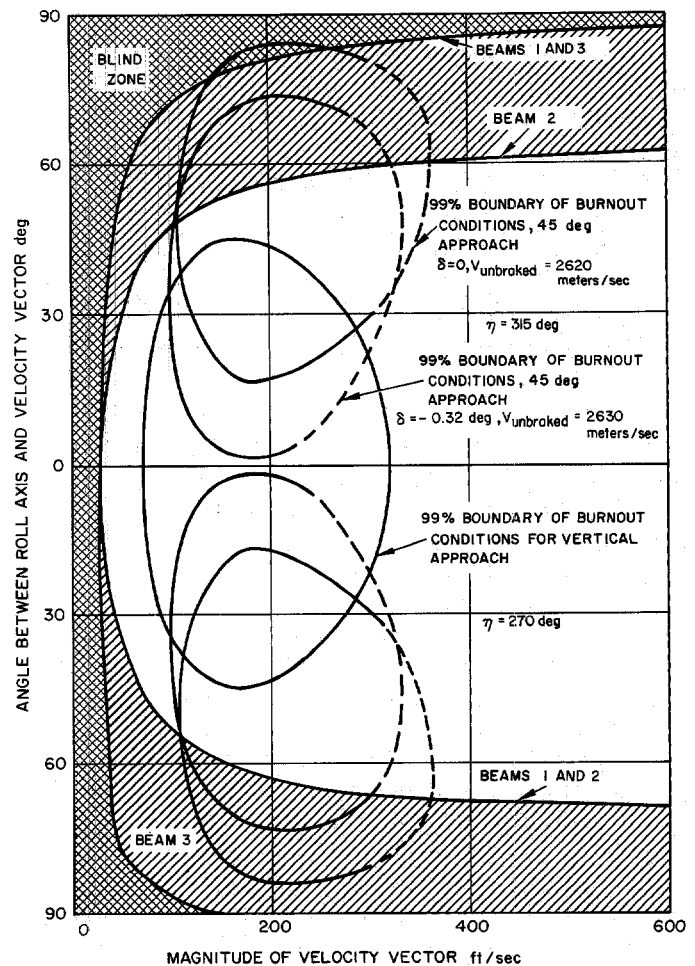


Figure 25. Blind zones and burnout dispersions

2. Limitations on Spacecraft Attitude Imposed by Doppler Radars

The acquisition blind zone regions of the doppler radar discussed in the preceding bimonthly progress summary have been studied under the conditions of a 45-deg oblique approach geometry. A burnout velocity (magnitude and direction) dispersion has been calculated for the 45-deg approach and minimum allowable approach speed condition. This dispersion has been used to set the permissible blind zone and therefore define the required acquisition region for the doppler radar.

Under conditions of minimum burnout velocity, the altitude at start of doppler operation is always less than 30,000-ft and the doppler tracker search limit is the dominant restriction on low-velocity acquisition (Fig 25). The tracker search limit as a function of velocity and angle relative to the spacecraft is shown in Figure 25 as the cross-hatched blind zone area. Two roll orienta-

tions relative to the velocity are shown: $\eta = 315$ and 270 deg (η is the angle between the spacecraft x axis and the projection of the velocity vector into the x-y plane). For $\eta = 315$ deg, one doppler beam, the spacecraft roll axis, and the velocity all lie in the same plane. Also shown in Figure 25 are the 99% boundary of burnout conditions for vertical and 45-deg approaches. Two dispersion boundaries are shown for the 45-deg condition. The upper 99% contour results when the nominal thrust and initial velocity are exactly aligned, that is, no bias has been commanded into the desired thrust line ($\delta = 0$). For this condition, burnout can occur with an angle between the spacecraft roll axis and the velocity vector up to 84 deg with a corresponding angle between the in-plane beam and the velocity of 105 deg. An opening rate for one beam (Beam 2) would exist; in fact, for $\eta = 315$ deg any angle of the velocity vector greater than 65 deg from the roll axis will cause an opening rate to exist in one beam. For $\eta = 270$ deg two beams will have an opening rate for any velocity angle greater than 72 deg. Doppler acquisition with sufficiently large opening rates would be possible if the tracker search region were extended and blocking at zero doppler were added.

The double cross-hatched regions indicate conditions where all three doppler beams cannot acquire. These regions must be avoided and therefore the nominal burnout boundary was shifted to avoid the cusp at 78 deg. The shift was accomplished by adding a slight bias to the desired thrust line ($\delta = -0.32$ deg) so that during main retro the thrust action was at 44.68 deg with respect to the local vertical. The $\eta = 270$ deg condition is not as critical as far as the three-beam restriction is concerned, but does indicate that two beams can be in the blind zone region for the 45-deg approach.

Based on the biased thrust dispersion contour of Figure 25, a permissible blind zone and required acquisition region for the velocity sensor was derived as shown in Figure 26. No burnout conditions should exist (based on the 99% contours) which place the velocity vector more than 75 deg from the spacecraft roll axis. Since extended opening rate conditions will not be present, the tracker search region will not be expanded to cover opening rate conditions.

Operation in the permissible blind zone will be accomplished by modifying the reliable doppler logic such that any one of the three velocity beams can signify reliable velocity control operation. A synthetic zero velocity signal will be supplied for those beams where acquisition has not been achieved. Attitude control will be carried out with a small initial error in maneuver direc-

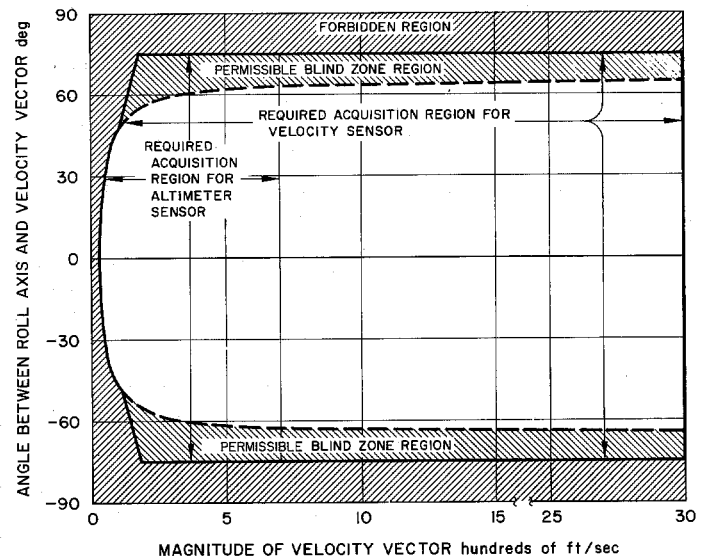


Figure 26. Required velocity sensor performance regions

tion (maximum of 35 deg) as described in the preceding bimonthly summary.

The emergency logic described above will be initiated only after a time delay sufficient to allow the tracker to search over its complete acquisition band. In addition, normal acquisition will reset the reliable operation logic to the present normal mode.

3. Transit Performance and Requirements as a Function of Injected Weight

The design of certain *Surveyor* modes of operation are directly dependent on injected weight while other modes are indirectly affected depending on the actual method of implementing the change in spacecraft weight. Certain functions or operations are essentially independent. However, for the critical weight-dependent modes of operation, such as main retro, vernier descent and touch-down, design changes can be made which ensure constant or improved performance as the injected weight is varied over some selected bounds. Certain ground rules were assumed in a study of large (500 lb) variations in injected weight:

- (1) The maximum and minimum thrust ratio of the vernier engine system would remain constant.
- (2) The final touchdown velocity and method of control (constant velocity to engine cutoff altitude) should remain essentially constant and within the present design specification.

(3) Vernier fuel could be off-loaded to achieve maximum payload. A vernier fuel reserve of 10 lb would be allocated, however, for configurations below 2300 lb.

(4) Minor flight control changes such as loop gain and maximum thrust descent parabola slopes would be allowed in order to accommodate variations in moment of inertia or achieve savings in vernier fuel.

(5) Velocity removed by the main retro would remain essentially constant; that is, main engine propellant is removed as spacecraft mass is decreased.

Using the predicted flight weight figures for the basic spacecraft, as given in Section A, and utilizing items 1 through 5 above, the required propellant, minimum touchdown weight, and resulting scientific payload is plotted as a function of injected weight in Figure 27. As reported in "Study of Lightweight Surveyor A," the selection of a specific payload for a reduced weight system resulted in an injected weight of 2099.3 lb. The corresponding propellant payload and touchdown weight is called out in Figure 27 for this injected weight. Ground rules 1 and 2 set the minimum injected weight that can

be considered. As touchdown weight is reduced, the minimum vernier thrust to lunar spacecraft weight approaches 1 lunar g, the limit for constant velocity descent. For the present vernier engine system, a minimum thrust of 90 lb (30 per engine) is possible, but 96 is the practical limit when attitude control requirements are considered. For a 96-lb lower thrust limit, 580 lb of touchdown weight becomes the lower limit. This corresponds to a lower limit of 2000 lb on injected weight.

To gain additional confidence in achieving cutoff altitude under the limiting lightweight conditions, two minor chances could be made:

(1) The constant velocity descent of 5 ft/sec could be increased slightly.

(2) The altitude at start of constant velocity could be decreased.

Figure 28 for the 2000-lb system shows the touchdown performance with an increase of constant velocity descent to 7 ft/sec and a decrease in altitude from 40 to 32 ft. As shown, a 10% error in commanded acceleration

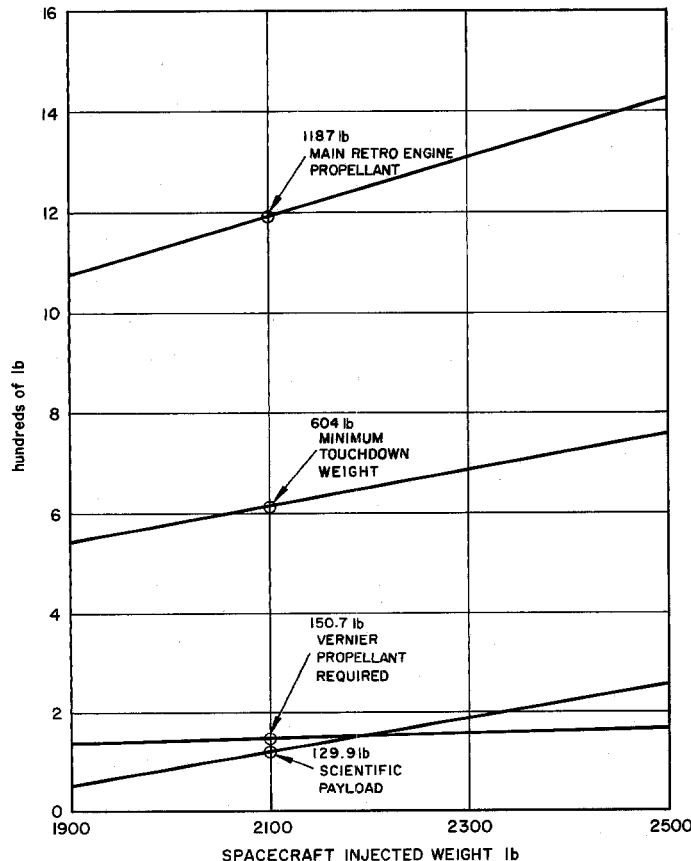


Figure 27. Effects of injected weight variation

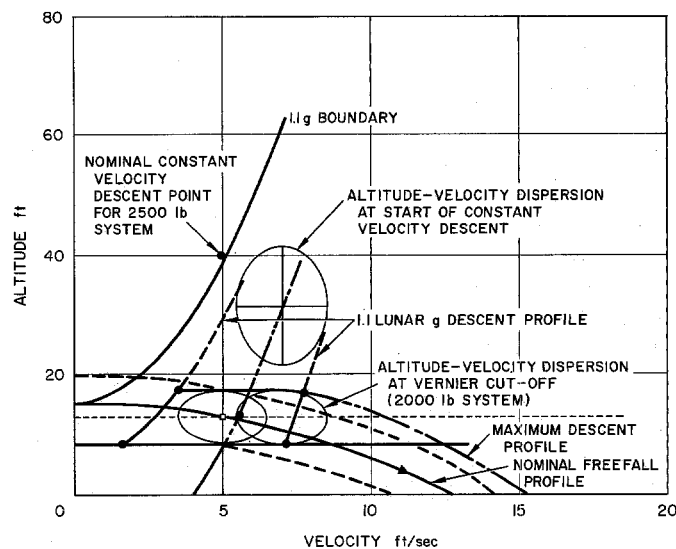


Figure 28. Touchdown velocity-altitude profile for spacecraft weight of 2000 lb injected, 580 lb landed

could be tolerated with this change, although the maximum touchdown velocity exceeds slightly 15 ft/sec. Figure 29 for a 2100-lb system utilizes only an altitude decrease (40 to 30 ft) but can still accommodate a 9% acceleration error.

Since the vernier system has continuous variable thrust control over a 3.46 to 1 range, it is possible to duplicate

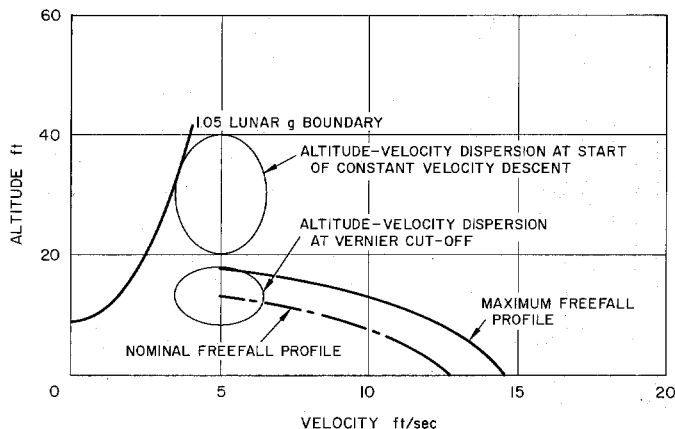


Figure 29. Touchdown velocity-altitude profile for spacecraft weight of 2100 lb injected, 604 lb landed

the 2500-lb maximum thrust descent altitude-velocity profile as the injected weight decreases. If no changes were made in the flight control descent parabola mechanization, a lighter weight system would follow the normal profile for the 2500-lb system. In this case, however, the average thrust commanded for a 2100-lb configuration is approximately 80% of maximum and nonoptimum vernier descent results. Fuel consumption is, of course, decreased but additional savings can be realized if the maximum thrust is utilized.

The 2100-lb configuration has a nominal high approach velocity start of vernier varying from 28,000 to 38,500 ft. In comparison, the 2500-lb configuration varies from 31,000 to 45,000 ft. More reliable operation of the doppler and altimeter sensors should be realized at the lower demanded altitudes of operation. In addition, the burnout dispersion in altitude and velocity are slightly reduced as the injected weight is decreased. This reduction is due primarily to the slightly decreased burning time and lower uncertainty of propellant burned, which results from a decrease in total propellant required. The

lower burnout velocity dispersion allows a slight decrease (≈ 10 ft/sec) in the minimum approach velocity. Because sensor altitude limitations and vernier fuel restrictions are relaxed for the lightweight descent profiles, the maximum approach velocity nominal burnout regions have been shifted 65 ft/sec higher. The two factors would increase launch window capability by increasing the allowed approach velocity variation by 75 ft/sec. Unbraked impact speed could thus vary from 2690 m/sec (8840 ft/sec) to 2612 m/sec (8560 ft/sec).

4. Engineering Instrumentation

Because of the emphasis on test and demonstration associated with lightweight configurations, a review of existing engineering instrumentation was made. Emphasis was placed on obtaining additional basic engineering data which is heavily dependent on relatively unknown environmental factors or system performance which cannot be accurately simulated on Earth.

Based on the above ground rule, the terminal phase up to successful touchdown is certainly the most critical. Sensor performance which is designed for wide variation in reflection is still predicated on a relatively unknown lower limit. Engine performance cannot be checked in a true vacuum and the craft lands on unknown surface conditions with a velocity that is dependent on the total past performance of retro, vernier, and flight control. Therefore, to gain as much information as possible, complete leg and load point instrumentation is indicated. Since relatively high-frequency information is desired and high-gain antenna pointing cannot be guaranteed, tape recording of the information is necessary.

Additional engineering instrumentation appears desirable in the areas of transmitter power, temperature, battery condition, vernier system manifold pressure, command verification, and mode conditions. A complete list of instrumentation points with additions proposed is indicated in Figure 2 in Section A.

D. Flight Control

1. Introduction

The flight control subsystem consists of those elements of the spacecraft which provide velocity and attitude control during the transit phase of the *Surveyor* mission. Functionally, the lunar radar and propulsion are included; both are covered more fully in other sections. Elements of the flight control subsystem which are treated in this section include optical sensors, inertial instruments, attitude controls, and flight control electronics.

2. System Design

a. Summary. During the reporting period, revisions in the control circuitry were incorporated and released to accommodate all outstanding system design modifications. These changes included the vernier retro phase acceleration command servo and the proportional servo actuator for thrust phase roll control, as well as some minor changes and improvements affecting the gas jet control circuits, the automatic logic circuits, and the vernier descent control law circuits. In addition, changes were made in the logic circuits of the flight control programmer to correct a condition involving the command sequence which would have required the radio transmission of the "Inertial Mode" command as a reset between many of the required transit modes.

Analog computer studies are continuing on the vernier phase attitude and velocity control system, the

coast phase, attitude control system, the thrust phase roll control system, and the T-2 vehicle dynamics. Results of some of these studies are presented in following paragraphs.

b. Coast phase attitude control. Optimization of the system parameters has been continuing with the aid of an analog computer. Adjustments have been made in some parameters to achieve better damping of angular motions in the acquisition and reorientation modes and to minimize the impulse required. A block diagram of the three-axis coast phase attitude control system is shown in Figure 30.

In the optical mode the primary damping signals are generated by the caged gyro loops. The loop gains K_1K_2 are chosen to be only large enough to prevent gyro saturation during initial stabilization. The optical channel gain, K_0 , was chosen to provide a deadband of 0.2 deg about each axis. The remaining parameter K_1K_4 has been selected on the basis of a parametric study of optical mode settling transients. The particular initial conditions used for these transient responses were chosen because they represent typical conditions encountered in Sun and star acquisition. The impulse and the settling time for each of the three control axes are shown in Figures 31 through 33.

As mentioned in the previous bimonthly progress summary, the effective deadband in the inertial mode is a

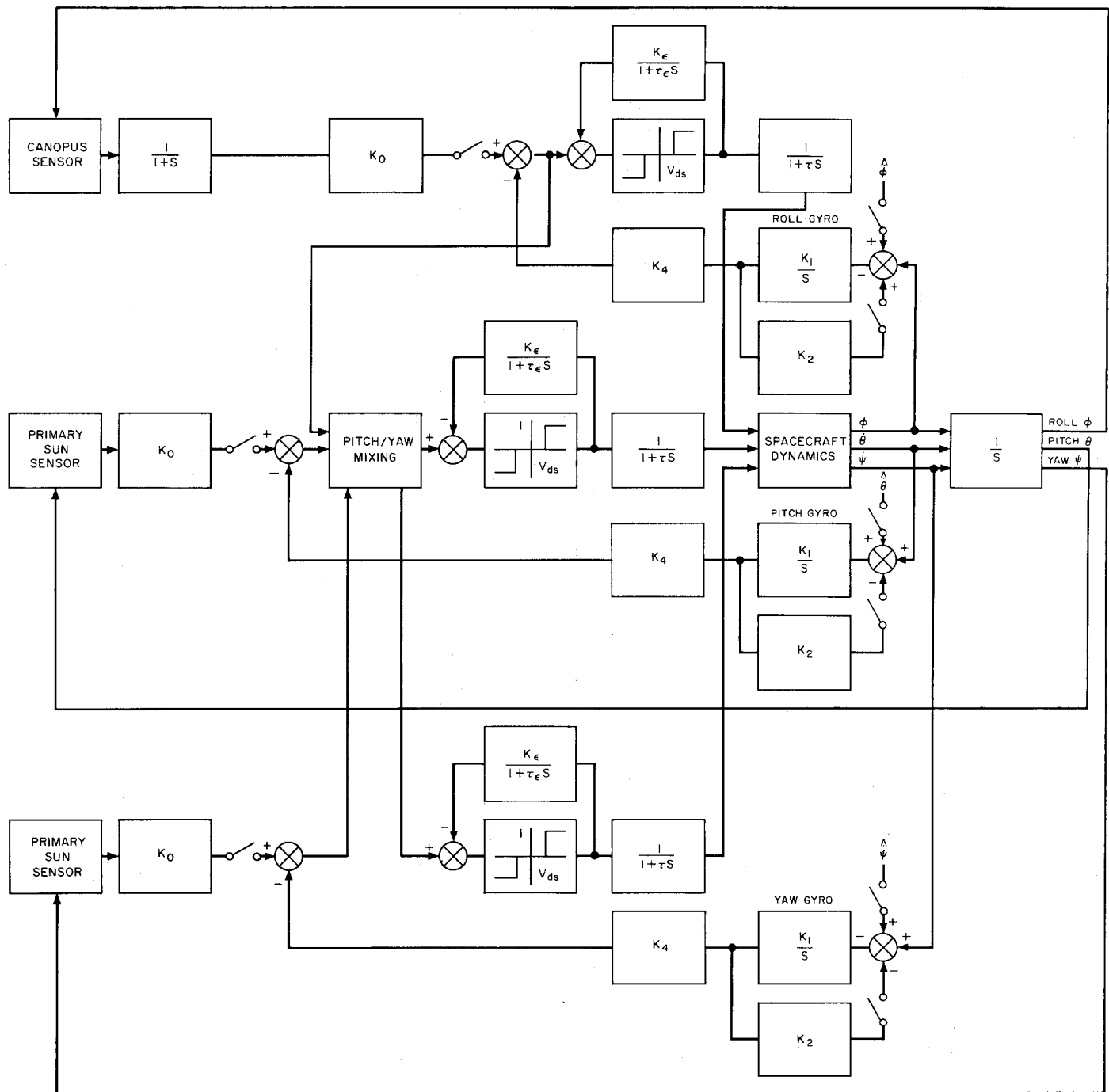


Figure 30. Coast phase attitude control diagram

function of the system damping in the optical mode once the gyro loop gain $K_I K_d$ is fixed. It is desirable to make the inertial mode deadband as wide as possible within the ± 0.2 deg limits established by pointing accuracy requirements. The graphs illustrate that a trade-off between inertial mode deadband and optical mode damping must be made. This tradeoff results in an in-

ertial mode deadband of ± 0.10 deg in roll, ± 0.0835 deg in pitch, and 0.129 deg in yaw.

For these deadband values, the dependence of inertial mode response (for a step rate command of 0.5 deg/sec) on the inertial mode damping parameters K_ϵ and τ_ϵ is shown in Figures 34 through 36. In these graphs, settling time is defined as the time for the angle between the inte-

grated rate command and the actual integrated rate to reduce to less than 0.2 deg. Based on these results, the values chosen for the inertial mode damping parameters are $K_e = 6 V_{ds}$ and $\tau_e = 3$ sec.

The settling time and impulse required in the inertial mode (reorientation maneuver) are dependent on the total angle commanded because of the interaction between the starting and stopping transients. For the selected values of K_e and τ_e , the inertial mode performance as a function of reorientation angle commanded is shown in Figures 37 through 39. These graphs illustrate that for attitude maneuvers in the range of 5 to 10 deg the settling time and impulse required may be roughly twice that for larger angles.

Acquisition of the Sun from a random orientation involves a sequence of rotations in the inertial mode, first about the yaw axis until the Sun line lies approximately in the roll-yaw plane, and then about the pitch axis until the primary Sun sensor is illuminated. The acquisition mechanics may be viewed in terms of the coordinate system shown in Figure 40.

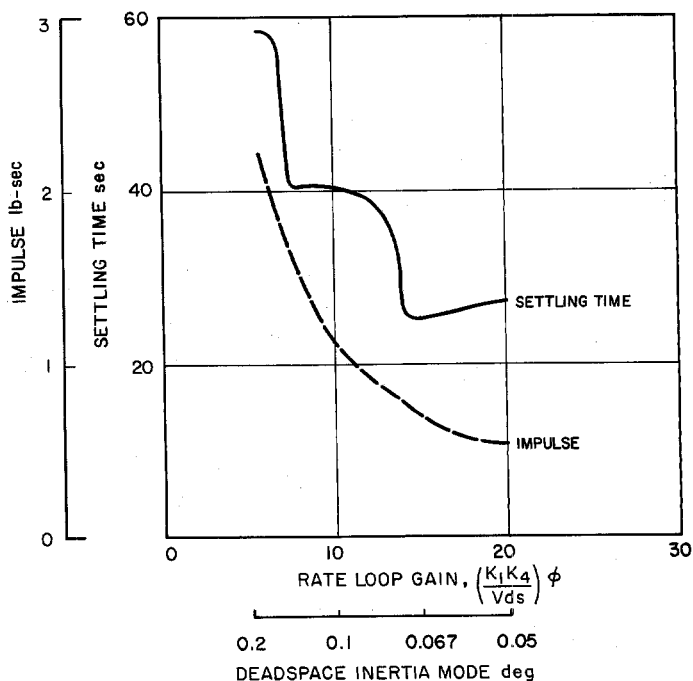


Figure 31. Roll optical control performance

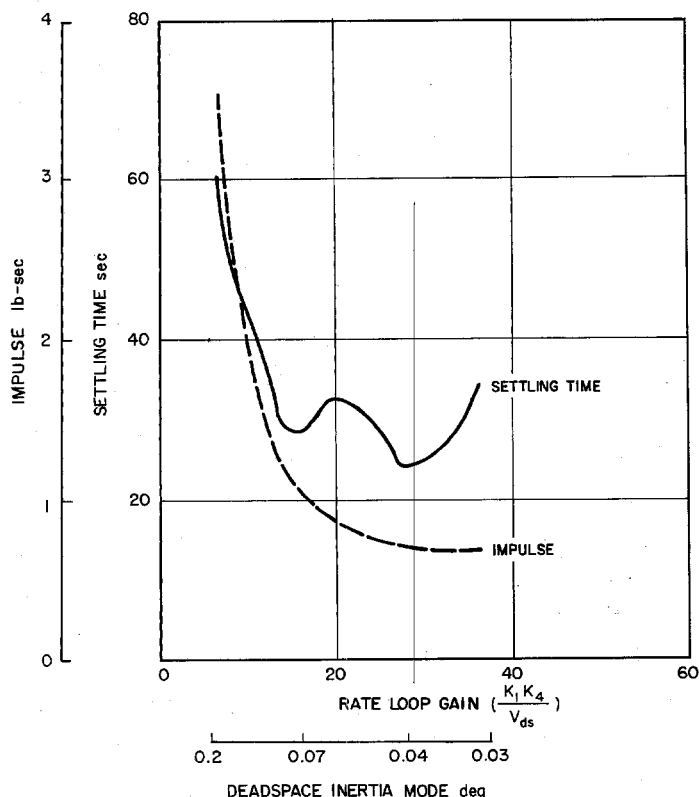


Figure 32. Pitch optical control performance

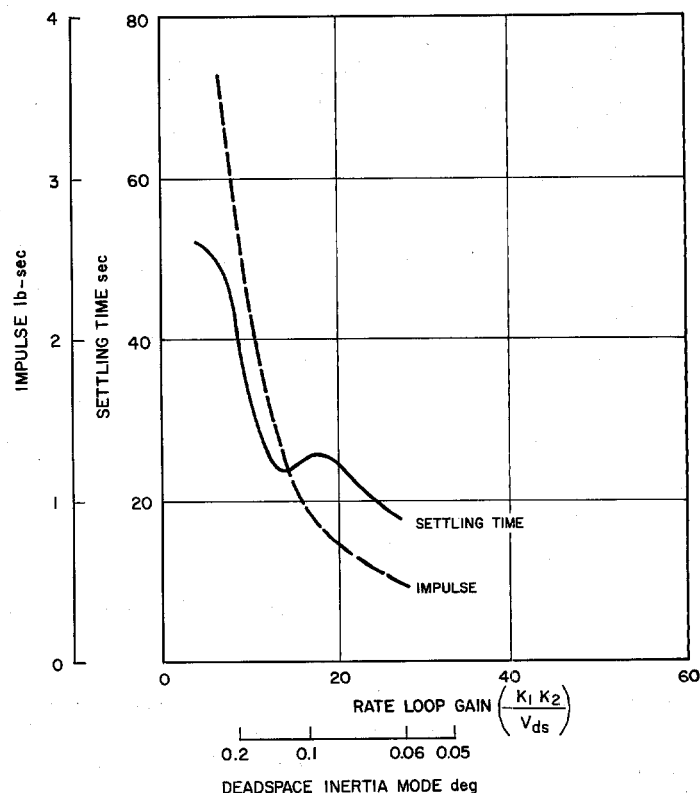
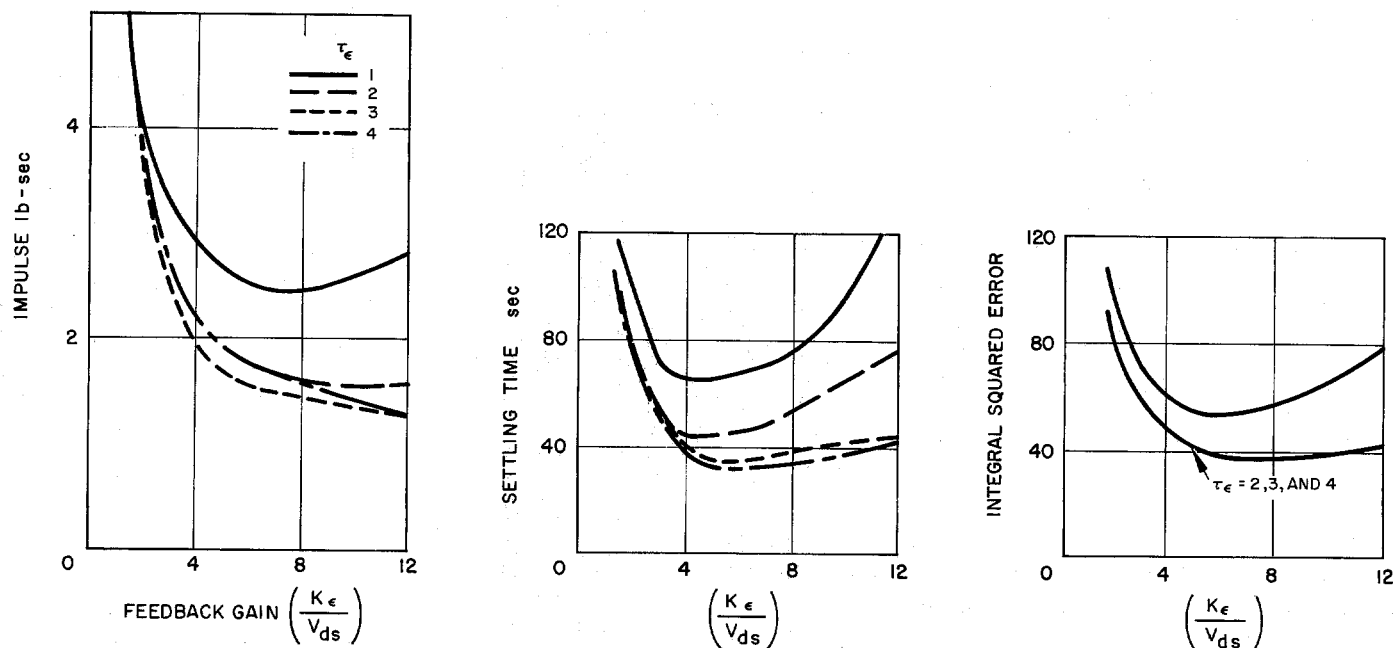
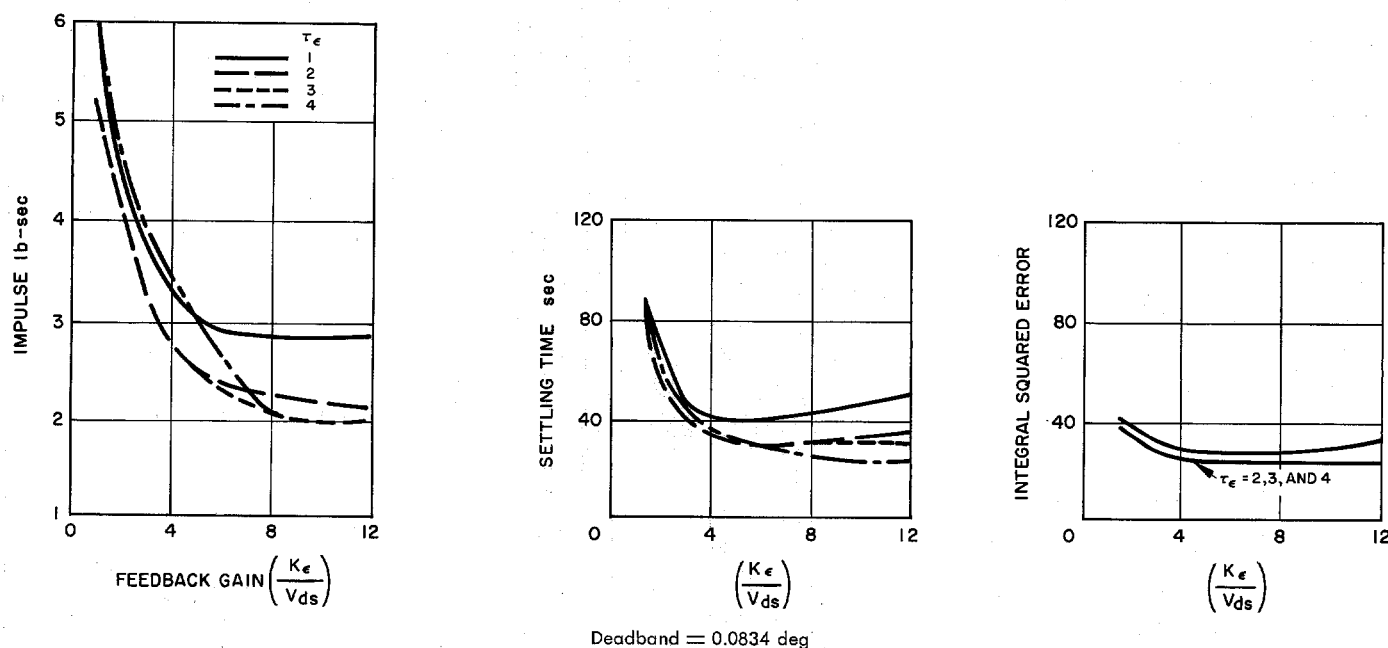


Figure 33. Yaw optical control performance



Deadband = 0.101 deg

Figure 34. Roll lag feedback parameter optimization



Deadband = 0.0834 deg

Figure 35. Pitch lag feedback parameter optimization

The outputs of the secondary Sun sensor are logically processed so as to produce the rate commands for the pitch and yaw loops as a function of Sun line position, as illustrated on the θ_p, θ_y space shown in Figure 41. When the Sun line moves within 8 deg of the roll axis, control is switched to the optical mode in both pitch

and yaw channels and simultaneous nulling of the pitch and yaw error angles takes place.

A typical trajectory in the θ_p, θ_y space is shown in Figure 41. Initially with $|\theta_y| > 1$ deg the spacecraft is given a yaw command rate of 0.5 deg/sec in a direction

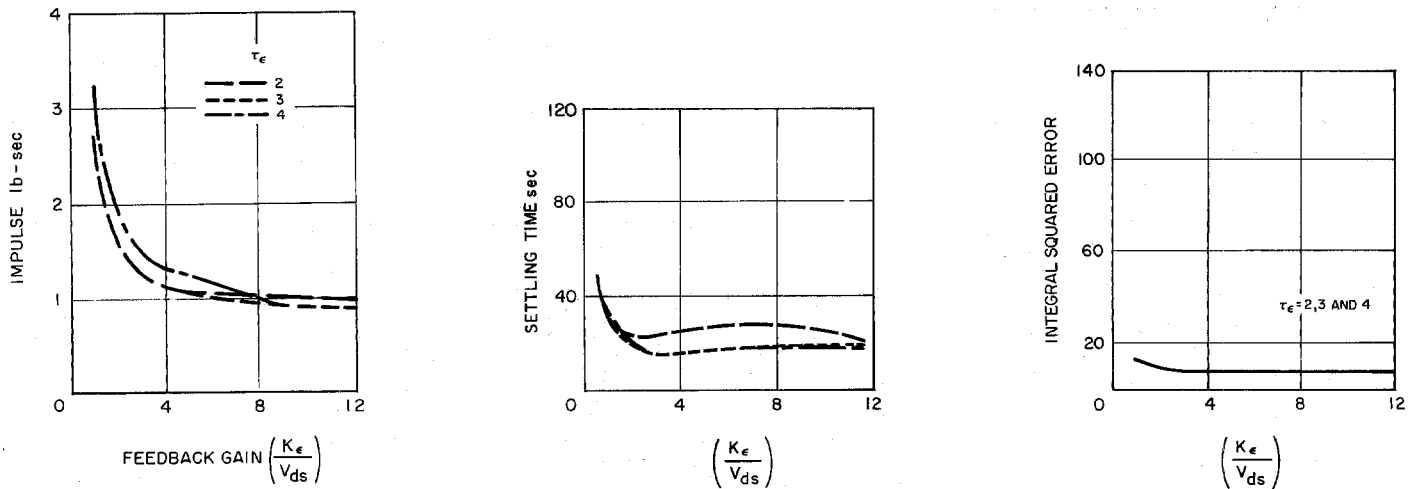


Figure 36. Yaw lag feedback parameter optimization

to null the angle. At the $|\theta_y| = 1$ deg boundary, the yaw command rate is set to zero and a 0.5 deg/sec pitch command rate is applied to reduce θ_p . The settling transient in the yaw channel involves a damped oscillation about the $|\theta_y| = 1$ switching line. To avoid reverting back to the yaw command condition, when $|\theta_y| > 1$ deg, a hysteretic range is provided in the flip-flop operated by the secondary Sun sensor. The hysteresis is set for a range of ± 4 deg which is enough to encompass the settling transient.

For angles $|\theta_y| < 10$ deg at the initiation of the sun acquisition maneuver, the yaw gyro gimbal and the yaw rate will not have achieved their steady-state values at time of crossing the $|\theta_y| = 1$ deg switching boundary. Under these conditions, if the pitch command were applied, an undesirably large yaw transient would result due to the coupling caused by the interactions between

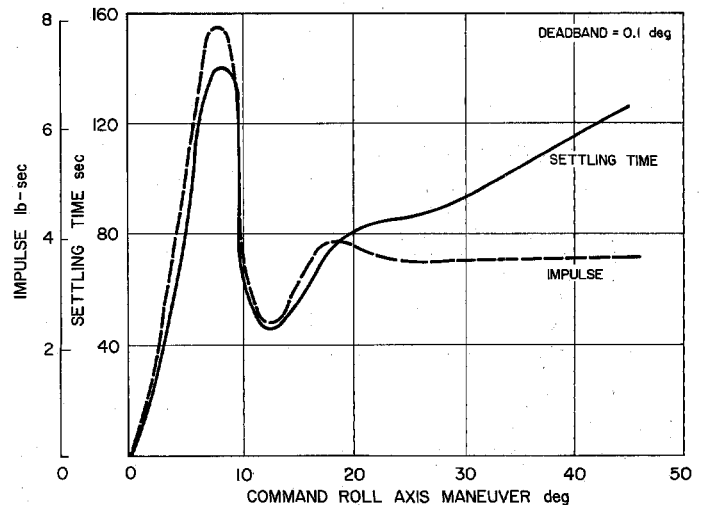


Figure 37. Roll inertial mode performance

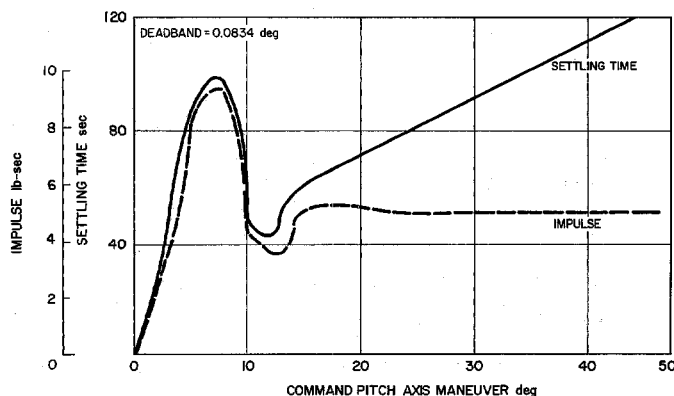


Figure 38. Pitch inertial mode performance

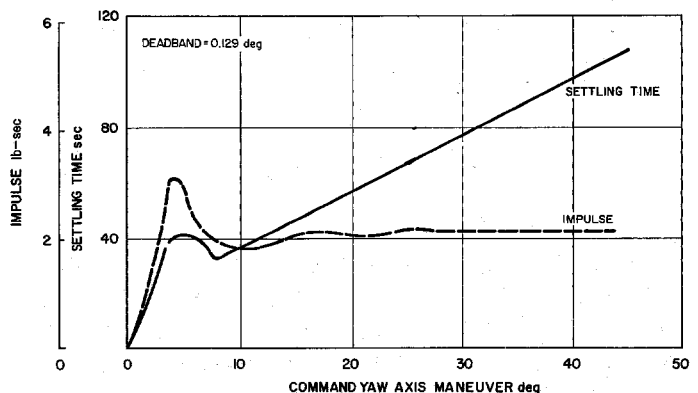


Figure 39. Yaw inertial mode performance

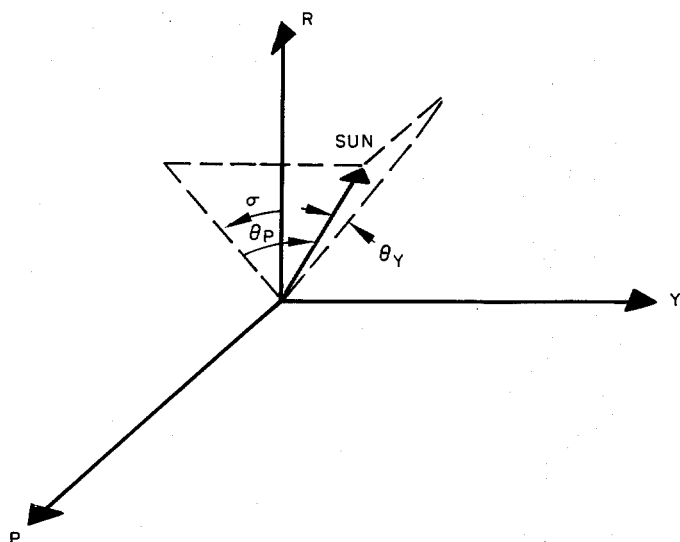


Figure 40. Sun acquisition reference coordinate system

the pitch and yaw jets. To avoid such a condition, a time delay has been introduced. This time delay prevents the pitch rate command from being executed until 32 sec have elapsed after the start of Sun acquisition. Therefore, for small initial angles of θ_y , the yaw settling transient is free to damp out before the pitch rate command is applied.

c. Vernier descent phase. The two areas currently under study in the vernier descent phase are as follows:

- (1) The optimization of the range-velocity profile.
- (2) The performance of the control system in terms of range, velocity, attitude, and attitude rate at vernier engine cutoff.

Figure 42 illustrates the basic form of the range-velocity profile. Four segments are used to approximate the desired command profile, two above and two below the 1000-ft range. Since it is now planned to switch the acceleration command limiter near mid-velocity on the descent profile, the reference curve will now consist of two parabolic segments defined by

$$\frac{dR}{dv} = \frac{V}{a_1} \text{ and } \frac{dR}{dv} = \frac{V}{a_2}$$

where a_1 and a_2 determine the thrust limiter settings. Optimum values of a_1 , a_2 , and the range at which the limiter is switched are being evaluated by an iterative procedure involving main retro burnout range, velocity, mass dispersions, and computations of vernier fuel consumption. The present mechanization is based on earlier

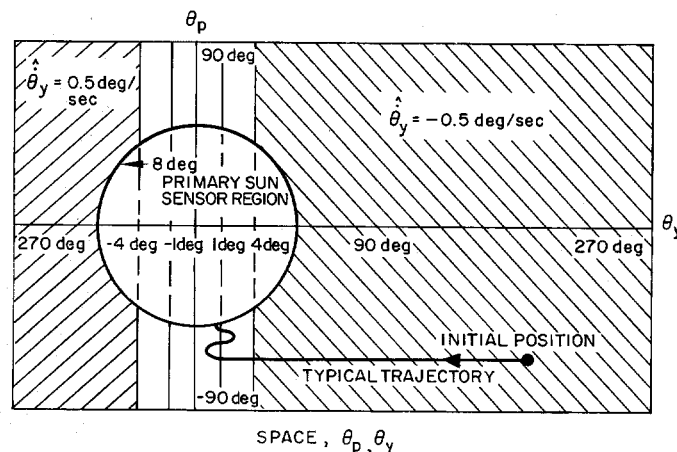


Figure 41. Typical Sun acquisition trajectory

approximations which are believed valid to within 10%. Flexibility in the circuit design has been provided to accommodate up to 30% variation in the parameters without affecting circuit layouts.

Those factors influencing the vernier engine cutoff variables may be considered in two categories:

- (1) Static factors: radar and engine alignment errors, electronic biases.
- (2) Dynamic factors: radar noise, nontracking vernier engines.

The hardware characteristics required to analyze the effects of these factors are being accumulated. The static factors can be treated for the most part by ana-

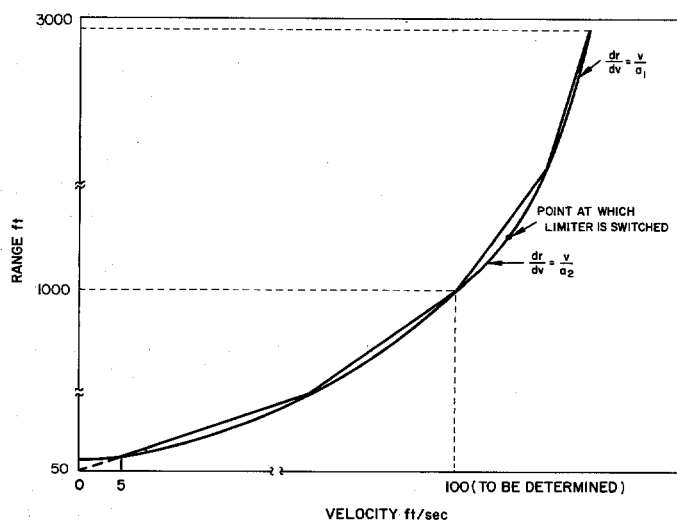


Figure 42. Range-velocity profile

lytical techniques. The dynamic factors on the other hand are being studied with the aid of a planar model analog computer simulation.

The effect of nontracking engines is to generate moment disturbances whenever a change in thrust command is encountered. A two-engine model, as shown in Figure 43, has been employed in the analog simulation to determine the magnitude of the transient. Such transients will occur at the time a straight line segment of the command range-velocity profile is acquired, at the end points of each segment, and when the final descent velocity of 5 ft/sec is commanded. The moment disturbances created by the acquisition of the lowest line segment and the constant velocity command are of particular interest in the cutoff dispersion study. Figure 44 is the recording of a computer run of the vernier descent trajectory in the range from about 1000

ft to vernier engine cutoff. The effects of nontracking engines during this trajectory on spacecraft attitude and angular rate are shown in Figure 45. The difference in thrust versus command slopes of the two engines in this run were 10%, which is estimated to be the maximum unbalance that may be expected.

d. Thrust phase roll control. The thrust phase roll control system has been redesigned to incorporate the proportional-positioning servo. The analog computer simulation, shown functionally in Figure 46, is being used to further investigate the system stability characteristics and performance capability. The design objectives are unchanged from those of the original "bang-bang" system and the performance criteria are still dictated by the maximum allowable values of roll rate ($\dot{\phi}$) and roll angle (ϕ) resulting from the various possible disturbance torques (L_d).

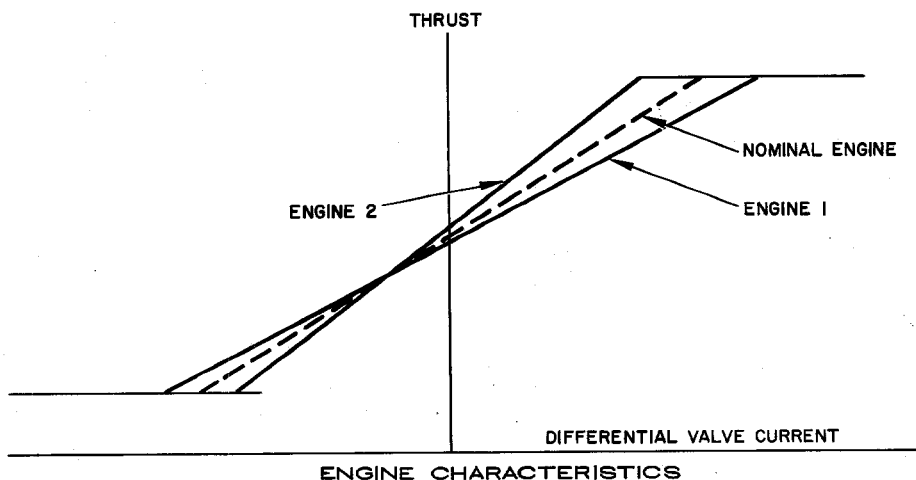
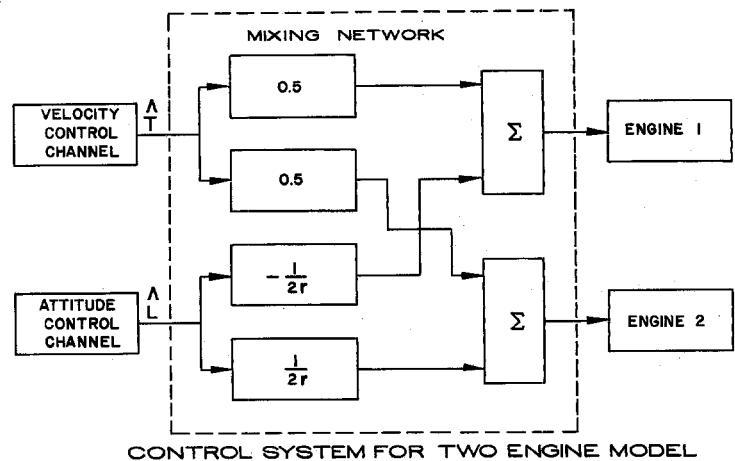
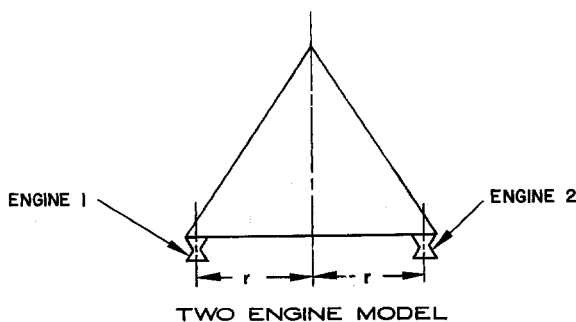


Figure 43. Nontracking engine model

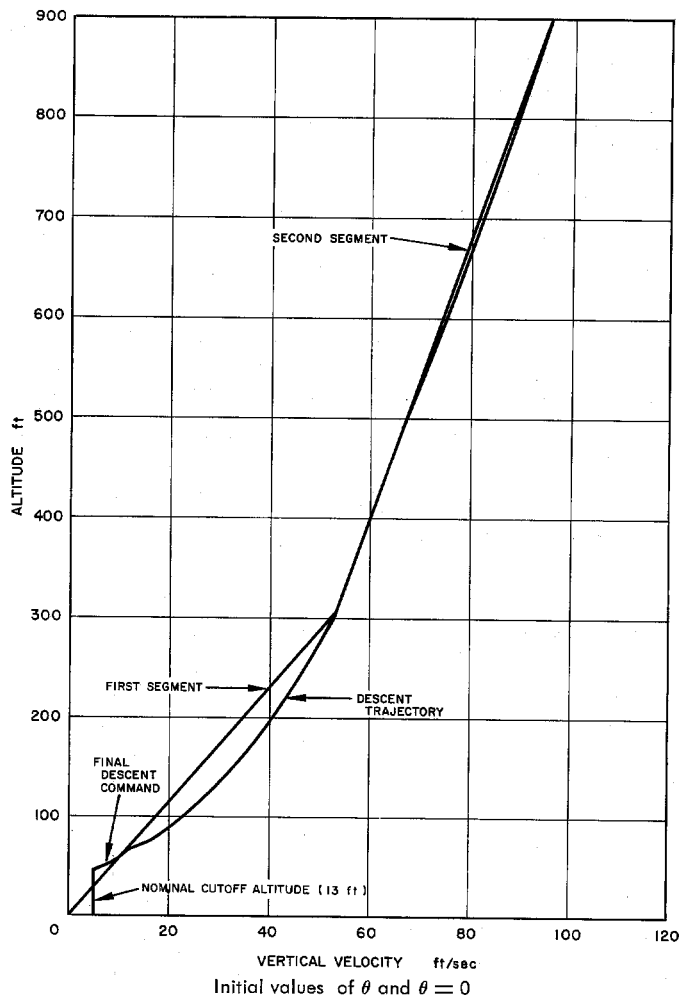


Figure 44. Vernier descent trajectory below 1000 ft

Command signal shaping and amplification requirements are being established to permit effective gain changes due to variations in vernier engine thrust level (T) and decreasing craft inertia (I_z).

The effects of the following system component characteristics are being evaluated and the component requirements established:

- (1) Command signal quality and limiting.
- (2) Servo velocity limiting.
- (3) Vernier engine angle limiting.
- (4) Backlash and friction of the vernier engine positioning drive-line.

3. Subsystem Tests

a. Single-axis air-bearing table. All breadboard units and associated hardware have been received for the

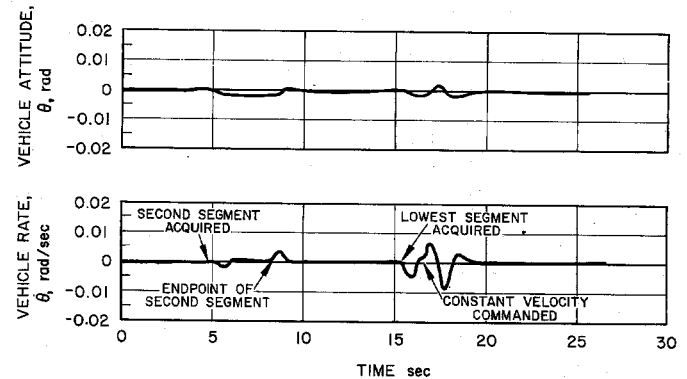


Figure 45. Attitude and angular rate during descent for nontracking engines

single-axis air-bearing table and table assembly has been completed. A functional checkout of the table and test console is planned following completion of table inertia and drift measurements.

A servo rate table with 0.0002 deg/sec minimum speed capability and with excursion limits of ± 4 deg has been designed to investigate the performance of the gas jet attitude control system during limit cycle. An analog computer will be used to close the loop by simulating the spacecraft dynamic response characteristics. This table will supplement the single-axis air-bearing test program and provide a backup capability in the event that spurious torques on the single-axis air-bearing table are excessive for an accurate simulation of the limit cycle behavior.

b. Flight acceptance test program. The laboratory facilities for subsystem testing have been completed. For initial tests involving the single-axis air-bearing table, breadboard versions of a star and solar simulator will be used. Final production models of both will be available prior to completion of the single-axis test program which is scheduled to start in mid-February.

4. Sensor Group Assembly

a. Summary. The sensor group assembly has not changed basically from the configuration shown in the last report. The basic structure drawings have been completed and thermal and mechanical models are now being fabricated in preparation for an extensive series of environmental tests. A series of thermal analyses on the sensor group have been performed and indicate that thermal balance can be attained in the present design with the structure plate at approximately 25° F. It may be possible to eliminate thermal insulation between the Canopus sensor and the support structure,

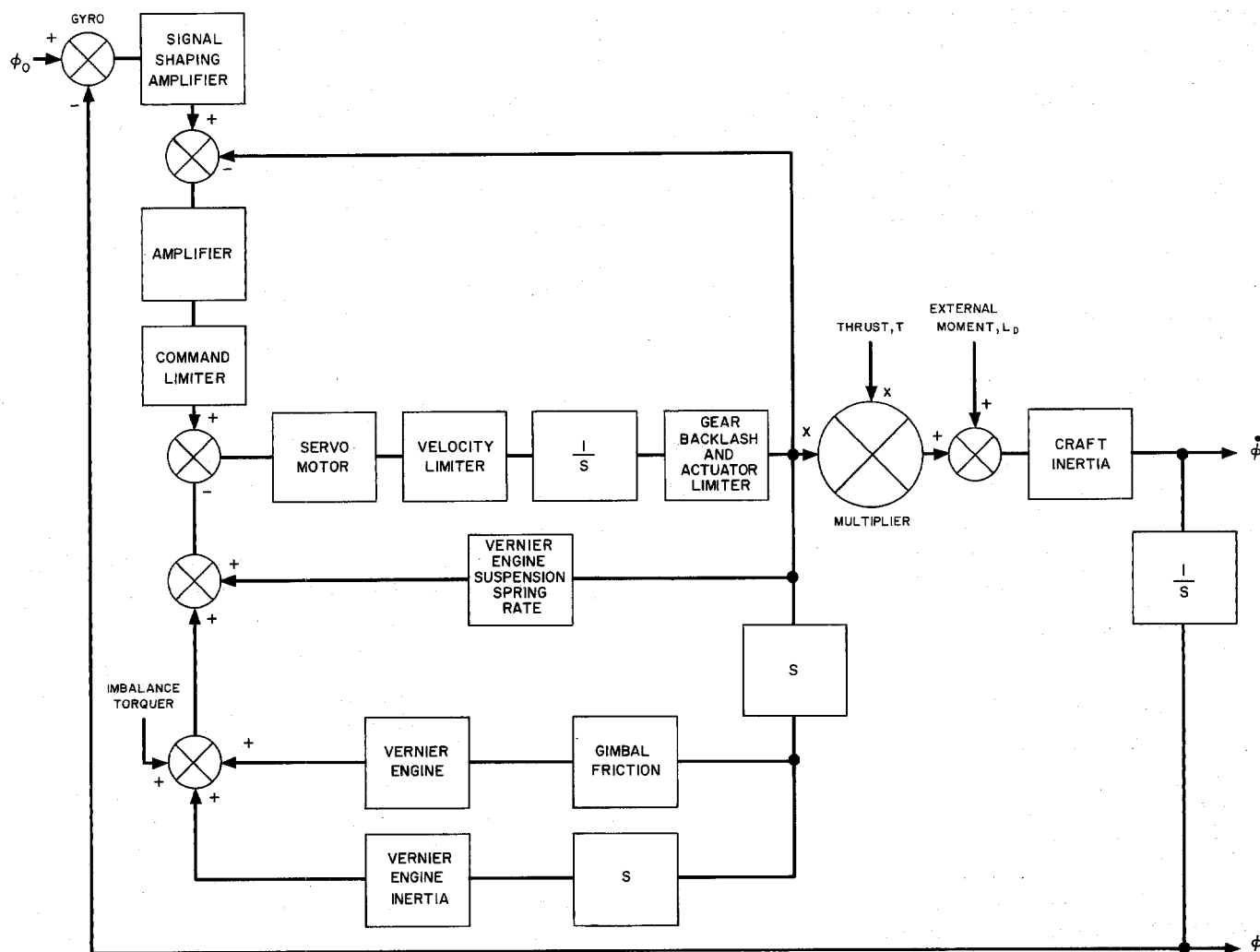


Figure 46. Thrust phase roll control system

although insulation between the electronics unit and the support will still be required. Surface finishes throughout will be a combination of polished aluminum and white inorganic paint with the exception of the inertial reference unit radiator which will be a second surface mirror. Detailed sensor group assembly procedures are being prepared. Initial emphasis is being given to aseptic sterilization procedures that may be necessary if the sensor group cannot be adequately tested after heat sterilization.

b. Electronics unit. All control circuit schematic drawings have been completed. Layout and assembly drawings for type approval hardware are being prepared. Figure 47 shows the chassis and connector arrangement within the electronics unit. Five chassis have been allo-

cated to control circuits while two chassis will contain electrical conversion unit and power switching circuitry, and three chassis will contain the flight control programmer and decoder circuits. It should be noted that each chassis has an individual connector for ease of checkout and maintenance.

Approximately four modules will be used on the chassis containing the control and power circuits, while the programmer and decoder circuits will consist of almost all modular-type packaging. All chassis will be of the standard foam sandwich type construction. In general, all large components will be mounted on the etched circuit board side. High heat dissipating components will be located on the chassis plate and as close as possible to a radiating surface. Heavier components will be located on the mounting side of the package in

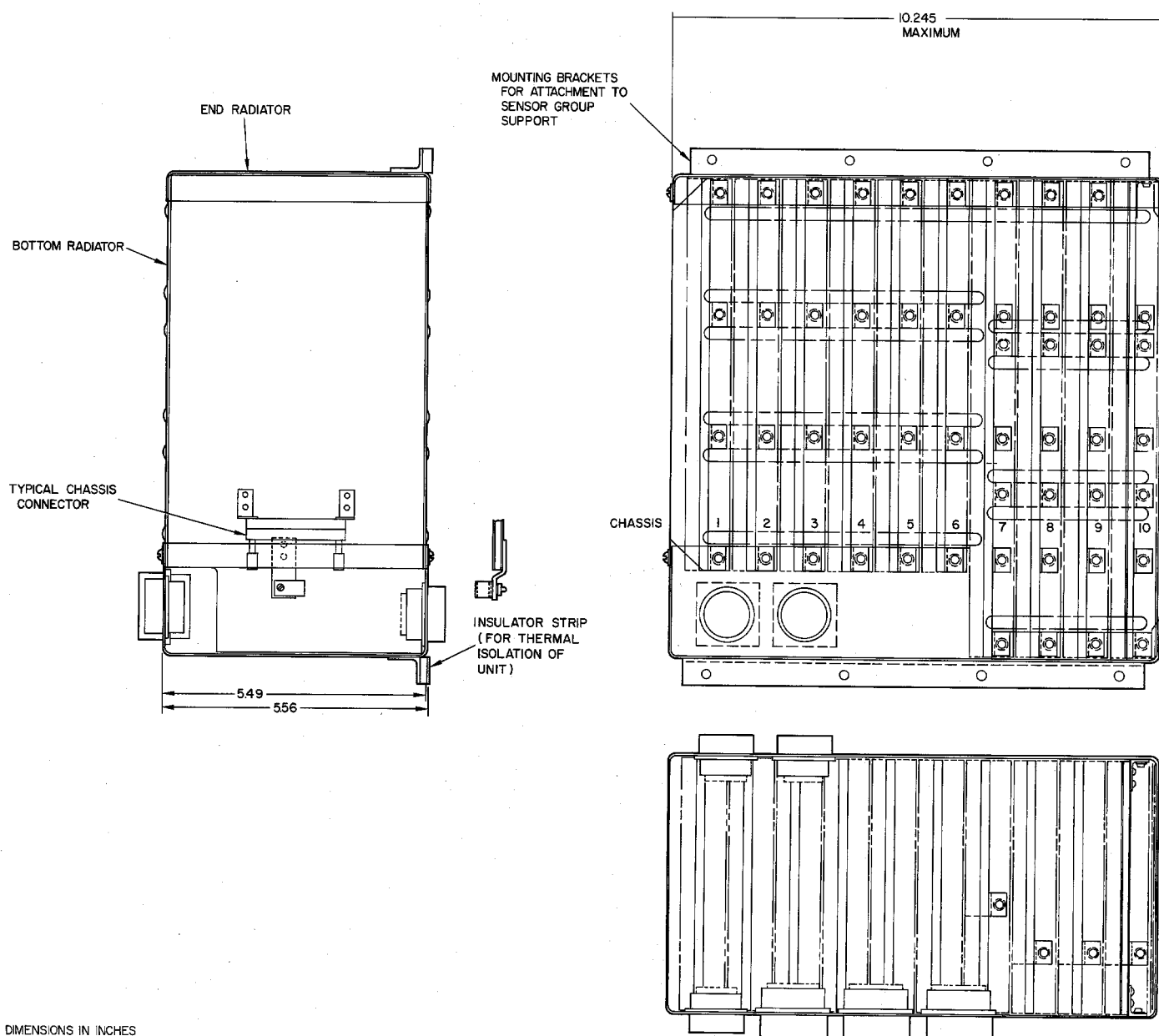
order to maintain the package center of gravity close to the mounting structure.

Results of a detailed thermal analysis indicate that only a portion of the bottom surface will be required as a heat radiator. Plans are to coat the bottom radiating surface to prevent radiation in excess of that required to obtain desired operating temperature during flight.

Drawings are almost complete and fabrication of the thermal mockup and mechanical unit will begin shortly.

Test equipment requirements have been ascertained and initial work has begun on fabrication of test consoles.

c. Inertia switch. Inertia Switch, Inc., was chosen as the vendor for the inertia switch. The inertia switch is a single-pole, single-throw, normally closed, acceleration-sensitive switch which is used to sense decay of the spacecraft rocket engine thrust below 3.5 g. This switch will be a part of the flight control sensor group package and, as such, will not require any alignment of its own to the spacecraft.



DIMENSIONS IN INCHES

Figure 47. Electronics unit chassis and connector layout

The switch that was selected is unusual in that the acceleration force on the seismic element of the switch is opposed by the magnetic force produced by an Alinco 5 magnet embedded in the mass. The g level actuation point is controlled coarsely by the strength of the magnet, and the fine adjustment is made by the adjustment of a soft iron slug at one end of the cylinder. This slug is then fixed in place with a set screw and epoxy cement. Viscous gas damping is used which is fixed by the clearance dimensions between the piston and the cylinder.

A balanced, multiple-wire, contact structure is used for the wiper, which makes intermittent switch operation very unlikely. The switch is simple (only one moving part and only a total of about ten parts), light, hermetically sealed, and it appears to be generally suited to *Surveyor* spacecraft operation.

5. Optical Sensors

a. Canopus sensor. The assembly of the development model of the Canopus sensor has been completed and a series of environmental tests on the unit have begun. A thermal mockup of the Canopus sensor has been fabricated and will be tested as part of the sensor group in spacecraft thermal tests. The mockup includes a 5-w heater to simulate power dissipation and two thermocouples to monitor the temperatures of the photomultiplier tube and the electronics package.

During the reporting period a study was completed on the S-4 magnitude of the southern star Canopus, as correlated with 23 bright stars in the northern hemisphere. The results of the study have been used to calibrate a Sun-star simulator which supplies appropriate optical inputs to establish correct settings of the Canopus sensor threshold discrimination circuits. Some study has been given to the effectiveness of the sky-mapping capability of the Canopus sensor. It is intended to telemeter the photomultiplier tube output as the spacecraft rolls through 360 deg. The present design has a detection sensitivity for stars of second magnitude and brighter. This sensitivity allows a 90% probability that a single star other than Canopus will be detected in a 360-deg sweep of the sky. Consideration had been given to increasing the sensitivity of the Canopus sensor to detect third-magnitude stars by incorporating a shutter which remove the sunlight amplitude reference from the photomultiplier tube. However, it is believed that the present design provides adequate performance and that the additional complexity required to increase the sensitivity is not justified.

b. Sun sensors. Tests are continuing on the developmental model of the primary Sun sensor. Minor modifications have been made as a result of these tests, and drawings are being prepared for the manufacture of type approval hardware.

Tests on the developmental model of the secondary sensor have also been conducted with various shadow mask configurations. The present configuration is shown in Figure 48. The orientation of the four acquisition cells is shown in Figure 49. Switching angles for the second-

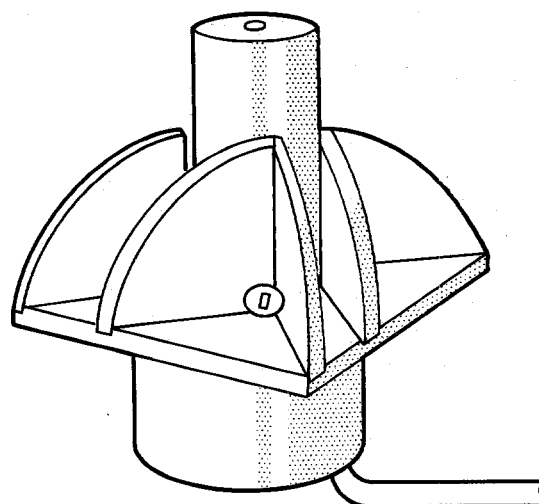


Figure 48. Secondary Sun sensor

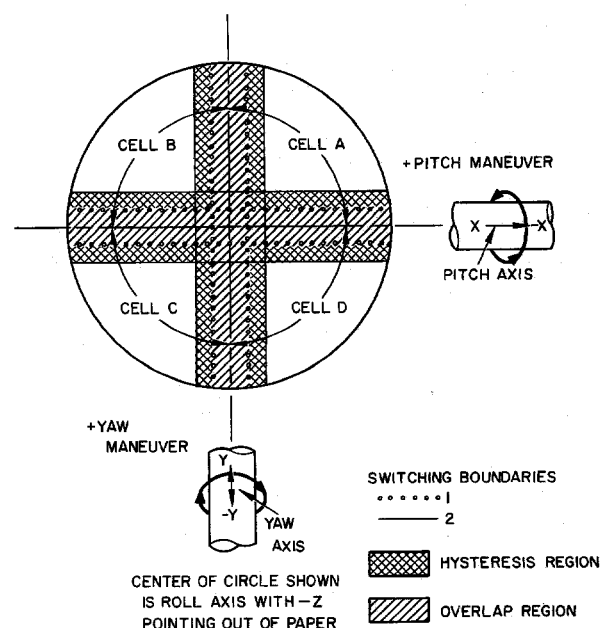


Figure 49. Secondary Sun sensor cell orientation

ary sensor are defined as the angle between the Sun line of sight and its projection on the roll-yaw or roll-pitch plane. The present sensor design has a resistance versus angle characteristic which allows the logic circuits in the control electronics to switch at an overlap angle of 1 deg (switching boundary 1) when moving toward the quadrant boundary and to switch at 4 deg (switching boundary 2) when moving away from the quadrant boundary. This hysteretic type of switching is a desired feature in the Sun acquisition mode, as described in 2. *System Design*.

A center "fifth" cell has been added with a narrow conical field of view (2 deg). The cell is connected in a resistance network with 5 v applied and its signal is used only to provide telemetered information to Earth. The signal peaks when the sensor boresight axis is pointed at the Sun and will be used to aid in determining the orientation of the vehicle on the lunar surface.

Tests are continuing on the Clairex photoconductive cells. A procurement specification on the desired cadmium-sulphide photoconductive cell has been sent out to several vendors including Clairex. Replies based on the specification will form the basis for the selection of the cells for the production Sun sensors.

6. Inertial Reference Unit

a. Instruments. An environmental test program for the *Surveyor* gyro has been received from Kearfott and evaluated as to adequacy and completeness. As a result of this evaluation, several studies were initiated. A study was made to determine the feasibility of performing both the gyro and accelerometer type approval tests at Hughes Aircraft Company. In addition, Kearfott was requested to reevaluate their type approval program based upon a modified program more consistent with the presently defined *Surveyor* environment. The quotation from Kearfott is expected soon, and a final decision will be made as to where the gyro and accelerometer environmental test programs will be conducted.

An acceptance test procedure and test data sheets have been prepared for acceptance testing of gyros. During this period, the gyro acceptance test console has been under fabrication and checkout in conjunction with a consignment gyro obtained from Kearfott pending receipt of the first production gyro. The development of the test procedures and test equipment proceeded at a somewhat slower pace than was originally estimated. The consignment gyro along with a thermally controlled fixture and support temperature control electronics were delivered to the single-axis test facility.

Final results of the accelerometer evaluation program led to the award of a subcontract to the Donner Scientific Corporation for a first buy quantity of seven accelerometers. First deliveries are presently scheduled for mid-February.

b. Unit design. The preliminary thermal and mechanical design of the reconfigured inertial reference unit structure has been completed and two units are in fabrication at this time. The mechanical design evolved into two basic structural elements: a glass cloth base epoxy laminate structure which provides the basic mounting and supporting structure, and an aluminum alloy weldment which provides the gyro mounting surfaces and the heat conduction-radiation link required for gyro temperature control.

The supporting structure is fabricated using the thermosetting plastic in the form of elements made from flat sheet stock bonded and screwed together to form the structure (Fig 50). Two major factors led to this

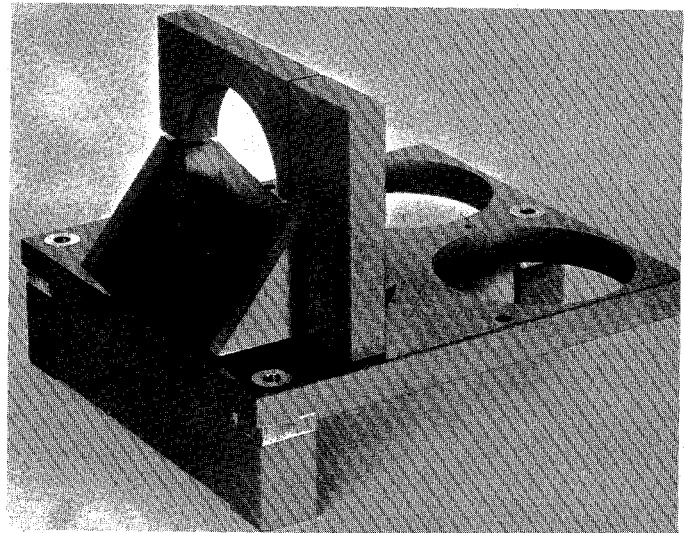


Figure 50. Engineering model of inertial reference unit support structure

choice of material. First, a material having a low thermal conductivity was required to minimize conduction heat losses to the sensor group structure; and second, the material had to have the physical properties to withstand the *Surveyor* shock, vibration, and cyclic temperature environments. Both thermal and stress analyses have indicated that all of these design goals will be met.

The thermal structure is a dip-brazed aluminum alloy subassembly connecting the gyro holders to the space radiator aluminum substratum (Fig 51). The factors gov-

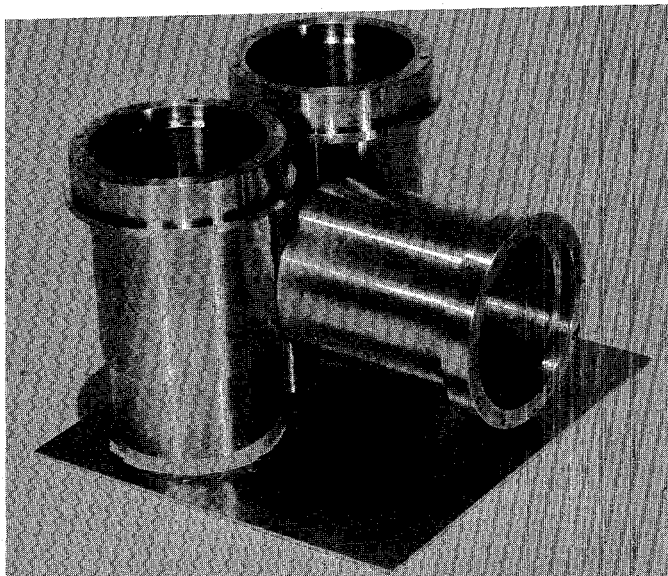


Figure 51. Engineering model of inertial reference unit thermal structure

erning the design of this element were predominantly thermal rather than structural; however, an analysis has shown that the element is more than adequate from the structural standpoint.

The new inertial reference unit design (Fig 52) has several distinct advantages. The use of a gyro temperature control amplifier for each gyro has eliminated the requirements to have equal conduction leakage paths to each gyro. Separate control loops will mean a modest reduction in average heater power requirements since the individual gyros have heat applied only when they require it and not as a result of interactions with the other gyros. The new space radiator orientation has allowed for optimum utilization of solar energy during transit phase, which reduces the amount of average thermal control power required. Finally, the plastic supporting structure yields a high thermal resistance to conduction losses to the sensor group structure.

Completion of the thermal design will be based upon the results of the thermal mockup evaluations to be conducted shortly on the entire spacecraft and radiation view factor studies to be accomplished on the full-scale mockup.

7. Attitude Control

a. Gas jet subsystem. Proposals received on the revised squib-operated valves are being evaluated. Additional bids may be required to locate a vendor who will

meet the design requirements and use the squib specified in Hughes Aircraft Specification 236182.

The jet valve has been changed to provide an AN pressure inlet connection in place of the long tube. Preliminary thrust-time measurements have indicated a more rigid test fixture will be required before accurate data can be obtained. The new fixture will sustain the shock due to solenoid armature hammer which was causing a noisy force transducer output.

The pressure control has been tested and appears to meet the predicted pressure discharge characteristics. Present test equipment is being modified to measure gas flow rate accurately. Minor hardware changes necessary to strengthen the discharge connection and increase its flow capability are being incorporated.

Test fixtures to test the high-pressure metal seals under vibration, acceleration, and shock are being made. Since actual tanks will not be available until later in the program, simulated titanium alloy surfaces will be incorporated in the test fixture. Also, since squib valve bodies and pressure transducers will not be available until later, the materials and surfaces of these units will be simulated.

Gas filter sterilizers are being investigated. Quotations and designs are being obtained on 0.3 μ filter elements which can be attached to the spacecraft and be heat-sterilized at Hughes Aircraft Company with the spacecraft. This equipment will be retained with the spacecraft until after gas charging. Spare sterilizing units are planned which will be heat-sterilized and available for

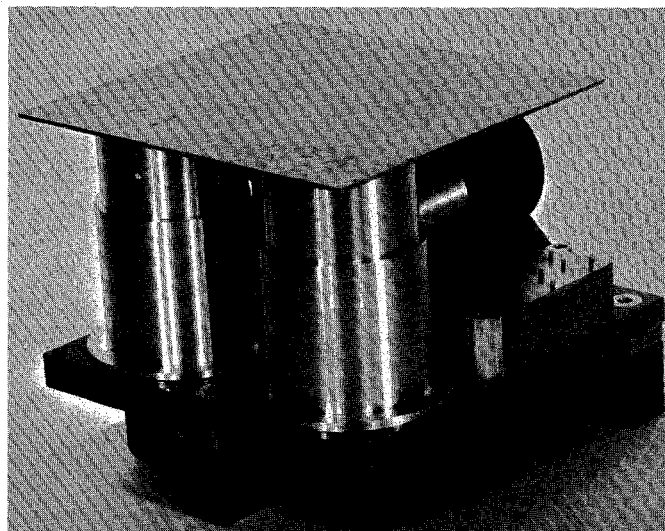


Figure 52. Engineering model of inertial reference unit assembly

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installation with aseptic sterilization if recharging at Atlantic Missile Range (AMR) becomes necessary.

Gas charging facilities are being investigated. One method being considered is the use of a typical gas pressure booster which will accept the AMR nitrogen supply. This unit would be used to charge a piston-type receiver which can be loaded with the correct gas weight and cooled sufficiently to prevent excessive charging temperature and pressure in the spacecraft tank when the gas is transferred.

A rigid and thorough quality assurance program has been planned to ensure the production of reliable gas tanks. Because of the limited production experience with the 7 AL-4Mo titanium alloy, additional provisions such as cyclic proof testing will be incorporated into the vendors' quality assurance program.

b. Roll actuator. The roll actuator as described in the last report period has now been defined in detail in Hughes Procurement Specification 234643. The specification has been submitted, with request for bids, to several companies who have considerable experience in the areas pertinent to the design and fabrication of this type of hardware. The technical proposals are presently being evaluated in preparation for selection of a vendor.

8. Control Circuitry

Revisions in the control circuit drawing release of November 1, 1961 were completed and re-released to the productization department on January 4, 1962. Besides the changes required for the acceleration command servo and proportional roll actuator which were discussed in the previous progress summary, the final release reflects minor changes to the gas jet control circuits, the logic circuits, and the vernier descent control law circuits.

Recent studies of the gas jet attitude control system have revealed the need for modification in the gas jet and gyro loop circuits. The reasons for these changes were discussed in 2. *System Design*. All of the changes will be accomplished without need for circuit relayout. An increase in dynamic range of the three amplifiers preceding the gas jet amplifiers is required with the new system parameters. The original design did not have the capabilities for the magnitude of the increase required. However, a solution has been found in the redesign of a transformer used in the circuit. Improved performance will also be realized in other circuits in which this transformer is used (i.e., the vernier engine attitude loops and the roll actuator servoamplifier).

An additional logic function called "precession defeat" has been included in the circuitry. This is utilized

to prevent an automatic reacquisition sequence from being followed in case the spacecraft attitude is disturbed during transit while telemetering is off. With this change the spacecraft will revert to inertial hold in its disturbed attitude until the condition is detected and reacquisition accomplished. The precession defeat mode also provides a ready solution to introducing the 32-sec time delay between initiation of the yaw maneuver and the pitch maneuver in the Sun acquisition sequence. Precession defeat as implemented in the flight control logic networks requires the addition of six components. However, a circuit improvement was also incorporated which resulted in the deletion of 34 components. Other changes in the logic required by the new Sun acquisition sequence described in 2. *System Design* included increasing the hysteresis of the flip-flops in the secondary Sun sensor circuits. This was accomplished by resistor value changes only.

The decision to use four straight line segments to approximate the parabolic descent trajectory was discussed in the previous Bimonthly Progress Summary. These four segments are generated in the diode-resistor circuitry shown in Figure 53. The complexity of this circuitry is the direct result of the following: the stringent tolerances imposed on matching the command to the desired trajectory; the uncertainty, system-wise, of the optimum trajectory; and the component parameter tolerances, especially under the influence of temperature.

The outputs of the doppler and slant range radars are both positive voltages with respect to ground, but they must be summed differentially to create a velocity error signal. The modulator (used as part of a low-drift chopper stabilized dc operational amplifier) is arranged to provide an ac signal with a peak-to-peak amplitude proportional to the difference between the two attenuated dc levels representing doppler velocity and slant range. The frequency of the modulator output is the same as that of the reference. When both dc levels are equal, the modulator output is at its null (less than 1 mv) and the current drain out of both radars is zero (the modulator has infinite input impedance). Thus, simple current summation, which assumes near-zero summing point impedance and nonzero current flow for null, is not applicable here. Voltage summation must be employed with passive attenuation used for scaling.

As shown in Figure 53, R5, R6, and R7 comprise the doppler attenuator, which provides a known attenuation against which the diode-resistor networks which produce the segments are scaled. R2 and R3 are chosen in conjunction with R1 and the parallel equivalent of R5, R6, and R7 such that the slope of segment number four is

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determined. The slope of segment 3 is provided by similar means except that R4 is also considered. R3 is returned to a precisely established and temperature compensated bias bleeder and provides the intercept of segment 4. The knee or transition between segments 3 and 4 is provided by the diode D1 coming out of forward conduction. Controlled lunar descent of the spacecraft starts on segment 1 and ends on segment 4; thus D1 is saturated and becomes unsaturated as slant range radar output voltage falls below the diode threshold which is also provided by the precise bleeder.

Segments 1 and 2 are established in the same manner; however, their circuitry operates on slant range radar outputs possessing a different scale factor. The scale factors are switched at 1000 ft for increased resolution, thus requiring a similar change in segment circuitry. At 10 ft/sec, the segments are switched out and a fixed command is substituted that provides a spacecraft deceleration equal to Moon gravity and a descent speed of 5 ft/sec until lunar touchdown.

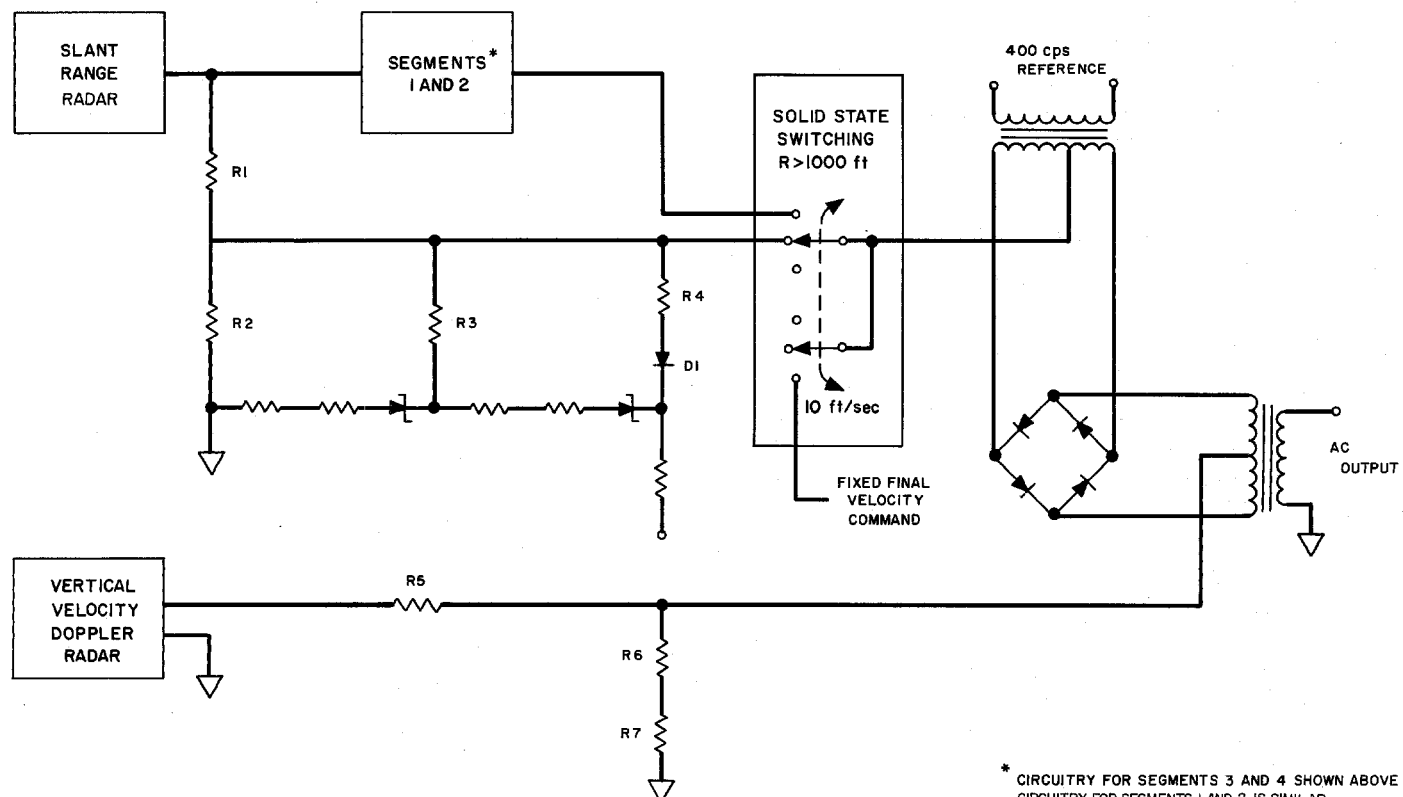
9. Flight Control Programmer

The flight control programmer, physically a part of flight control electronics, is a logic package which pro-

vides the flight control system with signals that control the operations of attitude control, midcourse velocity correction, main retro staging, nitrogen gate valve control, and engine thrust selection. Most of the inputs to the flight control programmer (FCP) are from the command decoder. Other inputs to the FCP are from the altitude marking radar, the legs-down switch, the retro separation sensor, and the inertia switch. These input signals when they arrive in certain predetermined sequences or combinations are enabled by FCP logic to generate yes/no output signals the state and duration of which control the flight operations.

The operations of the FCP are performed by flip-flops, diode gating circuits, and several types of logic amplifiers and inverters. These circuits may be grouped according to the functions of (1) memory, (2) time delay operation, (3) synchronizing, (4) input/output, and (5) internal logic decision circuits.

Memory is performed by a set of five mode function flip-flops and by a set of four "independent" flip-flops. The mode function (MF) flip-flops operate together to define 27 used states. Each MF used state corresponds to some function or mode of some function. The four independent flip-flops are each assigned a separate role.



* CIRCUITRY FOR SEGMENTS 3 AND 4 SHOWN ABOVE
CIRCUITRY FOR SEGMENTS 1 AND 2 IS SIMILAR

Figure 53. Descent control law trajectory circuitry

Table 8. Transitions involving transit modes of the FCP mode function flip-flops

From	To
1. Any state	Inertial — by inertial mode command
2. Inertial	Pitch to follow
3. Inertial	Roll to follow
4. Inertial	Rate mode
5. Inertial	Velocity correction to follow
6. Inertial	Yaw to follow
7. Pitch	Inertial (automatic)
8. Pitch to follow	Pitch
9. Rate	Rate and precession defeat
10. Rate and precession defeat	Sun and precession defeat and 32 sec
11. Roll	Inertial (automatic)
12. Roll to follow	Roll
13. Sun	Sun and manual star
14. Sun	Sun and pitch to follow
15. Sun	Sun and roll to follow
16. Sun	Sun and star
17. Sun	Sun and yaw to follow
18. Sun and manual star	Sun
19. Sun and manual star	Sun and manual star and roll to follow
20. Sun and manual star and roll to follow	Sun and roll
21. Sun and pitch to follow	Pitch
22. Sun and roll	Sun (automatic)
23. Sun and roll	Sun and manual star (with manual star command)
24. Sun and roll to follow	Sun and roll
25. Sun and star	Sun and manual star
26. Sun and star	Sun and star and precession defeat
27. Sun and star	Sun and star and roll to follow
28. Sun and precession defeat and 32 sec	Sun mode (automatic)
29. Sun and star and precession defeat	Sun and star and precession defeat and roll to follow
30. Sun and star and precession defeat and roll to follow	Sun and roll
31. Sun and yaw to follow	Yaw
32. Sun and star and roll to follow	Sun and roll
33. Velocity	Inertial (automatic)
34. Velocity to follow	Velocity
35. Yaw	Inertial (automatic)
36. Yaw to follow	Yaw

Table 9. Transitions involving main retro staging

From	To
1. Any state	Retro magnitude to follow (by retro magnitude to follow command)
2. Retro magnitude to follow	Retro delay τ_1
3. Retro delay	Delayed ignition and τ_2 (automatic)
4. Delayed ignition and 32 sec	End of acceleration time (automatic)
5. End of acceleration time	Burnout signal and τ_3 (by inertia)
6. Burnout signal and τ_3	Delayed burnout and τ_4 (automatic)
7. Delayed burnout and τ_4	Delayed — delayed burnout (automatic)
8. Any state	Delayed ignition and τ_2 (by emergency delayed ignition command)
9. Any state	Delayed burnout and τ_4 (by emergency retro eject command)

The transitions between MF used states are controlled principally by signals from the command decoder. Some transitions result from signals internal to the FCP. A list of transitions currently available is given in Tables 8 and 9.

Time delays are generated by transferring a 10-bit magnitude into a 10-bit register and then counting pulses from a reference clock source. During the time delay interval, the bits in the register count down by a circulation and subtraction process. Each occurrence of 10 clock pulses results in the number in the register being decreased by one count. The delay is ended when the register contains all zeros.

The number in the register may be obtained externally or internally to the FCP. The external numbers or magnitudes are variable and are transmitted from the ground after a magnitude-to-follow command. The internal magnitudes are fixed by FCP logic and stored in FCP decision elements.

Synchronizing circuits are required by the need for harmonious operation of FCP flip-flops in the presence of the following four separate sources of timing: command decoder shift clock pulses, command decoder command pulses, and two FCP reference clock frequencies of 25 cps and 200 cps. Synchronizing is performed by flip-flops and a one-shot integrating gate circuit.

Input/output buffering is accomplished by amplifiers and flip-flops designed to meet the needs of the FCP interface requirements.

Decision elements are resistor/diode gates of the AND and OR type. These gates are supplemented by amplifiers and inverters as required by the fanout of the logic and the loading capabilities of the circuits.

During this bimonthly interval, changes were made in the following areas of flight control programmer operation: main retro burnout recognition, vernier engine thrust selection, precession defeat, Sun acquisition delay, and use of the inertial mode.

a. Main retro burn-out switch interlock delay. The present concept for determining the point in time at which the main retro has burned out employs an inertia switch which is opened by the gradual acceleration after ignition and is closed by the relatively sharp deceleration after burnout. The opening of the switch does not supply any useful information but serves only to "set" the switch for its proper state prior to the sensing of retro burnout. The switch is normally closed when no acceleration is sensed. During the gradual acceleration of the spacecraft, the switch is expected to bounce for several seconds. To

prevent switch closure during this "noisy" acceleration interval from falsely indicating main retro burnout, it is necessary to provide an interlocking function such that no burnout indication can be sensed unless the switch has been fully opened and unless the acceleration has previously exceeded the switches critical actuation level in the neighborhood of 4 g.

To interlock this switch closure, a preset time delay of approximately 32 sec will be generated by FCP circuits following receipt of the main retro ignition signal. During this delay interval the FCP logic will be made insensitive to any signals from the inertia switch.

To implement this delay, the FCP logic has needed only minor changes because the basic ability of the FCP to generate internally defined time intervals has been used.

b. Vernier engine thrust selector. Two commands have been added to allow a selection of vernier engine thrust level during retro firing. The two commands are Nominal Thrust Bias and Thrust Bias Reset. The memory function for nominal thrust bias was accomplished by the addition of a flip-flop. Gating has been incorporated to control this flip-flop by the above two commands. Moreover, the resetting of this flip-flop may also be accomplished by the inertial mode command.

c. Precession defeat output. A new command and output have been added to cause the spacecraft to hold attitude by inertial reference in the event of loss of Sun or star lockon. The precession defeat command has been mechanized with certain states of the mode function flip-flops to result in a 10-v step from the precession defeat output circuit. No new flip-flops were added.

d. Sun acquisition delay. Recent design analysis indicated a potential problem in sun acquisition due to interaction between the yaw and pitch maneuvers. The solution to this problem was the addition of a delay of approximately 32 sec between execution of these two maneuvers in the automatic acquisition sequence.

The incorporation of this delay was simplified by time sharing logic circuits originally intended for the main retro sequence internally generated time delays.

e. Use of the inertial mode command. The previous logical design of the FCP was such that it was necessary to transmit the "inertial mode" command as a reset of mode function flip-flops to occur between any of the required transit modes. This was done to permit maximum operational flexibility with a minimum of decision elements. However, operation difficulties were discovered which, although not preventing mission success, did in-

crease the number of commands (up to $\approx 50\%$) and thereby made the missions operation job more difficult. Moreover, the previous design also required that the register, used to store the magnitude bits in accurate time interval generation for the precision attitude maneuvers, had to be set up while in inertial hold. Depending on the length of time required to transmit and verify the attitude maneuver information, this could have become a serious problem due to gyro drift.

Schedule and manpower were considered in detail to determine the implications of a redesign of the FCP to overcome these difficulties. The results of these considerations led to a decision to go ahead on the logic redesign.

The redesign had its principal effect on the operation of the mode function (MF) flip-flop state transitions. Complete flexibility would have required that if 27 states of the MF flip-flops were used, then a total capability of 702 transitions (i.e., 27 times 26) would be needed. Functional requirements of the FCP were reviewed, and 30 transitions were deemed sufficient for operation of the FCP in the transit modes. After the redesign effort had begun, an additional six transitions were added. The total set of 36 transitions during transit operation is shown in table 8. The MF flip-flops are also used during main retro staging. The nine transitions for this sequence are listed in table 9. The transitions listed in these tables are accomplished by appropriate radio commands or as indicated in the parentheses in the "To" columns.

The work on the logic design of the flight control programmer during the interval of this report was aimed at integrating the latest functional requirements into a completely revised and integrated set of logic equations. The new functional requirements incorporated in this design affect the following areas of operation: main retro burnout recognition; engine thrust selection; precession defeat; Sun acquisition delay; and use of the inertial mode. In spite of the extra functions added, the net part count for the new design is not expected to increase substantially. In fact, it is expected that a savings may be made in several logic amplifiers and flip-flops in the new design.

Circuits for the FCP are in the final design stage. Completion is scheduled for late February, 1962.

A breadboard of a 400-cps sync pulse generator has been delivered to flight control engineers. Two magnitude command simulators have been completed and are in checkout. A complete programmer breadboard is scheduled for completion in March.

E. Electronics

1. Introduction

This section covers the various systems of the spacecraft which provides the functions of telecommunications, decoding of Earth-originated commands, and conditioning and processing of spacecraft signals for telemetering to Earth. Included also are the radar systems associated with flight control.

Spacecraft flight control electronics, power management, and the command data handling console demodulator are discussed in Sections D, F, and L, respectively. The television and the tape recorder are discussed in Section B.

2. Telecommunications Decibel Control

The *Surveyor* spacecraft receives and transmits data to Earth in a variety of conditions. These conditions utilize the flexibility of the telecommunications link by changing such parameters as antenna gain, transmitter power, and data bandwidth. In order to provide assurance that the telecommunication link will adequately transmit and receive data under all conditions, the transmission equation has been calculated for each case. These calculations are documented in a *Surveyor* Laboratory Detail Specification which has been recently rewritten, and the results are reported here.

A typical calculation is listed in Table 10. This is for the transmission of TV data from the lunar surface. Sec-

ond, results of each calculation of the Detail Specification showing the expected signal-to-noise margin and the expected tolerance appear in Table 11. Telecommunications parameters which are not given in the lunar TV calculation but are used in other calculations are shown in Table 12.

a. Television transmission calculation. The parameters of this calculation and each calculation are combined in such a way as to determine the signal-to-noise ratio of the transmitted information and, by applying suitable thresholds, to determine transmission margins. For convenience the values of the parameters are given in terms of ten times their logarithm to the base ten with a sign which allows algebraic addition.

The center margin for the television, (Tab 10) taken with its tolerance, would allow a negative minimum margin. This result includes the demand that the signal-to-noise ratio (S/N) in the IF amplifier always be +12 db or greater. Actually, the desired results is an acceptable S/N in the detected video picture. This level of acceptability has been defined as 30 db S/N for the detected video in a 220 kc bandwidth. The relationship between the IF S/N and the video S/N for the *Surveyor* TV has been previously reported. The relationship given between the video S/N and the IF S/N was as follows:

$$(S/N)_v = 19.1 \text{ db} + (S/N)_{IF}$$

If the minimum S/N expected in the IF is taken from the previous calculation, this equation becomes:

Table 10. Television decibel margins

Parameter	Value	Tolerance
Transmitting power	10 dbw	1 db —0 db
Transmitter cable and switch loss	—0.4 db	
Interconnecting cable and transfer switch 1	—0.45 db (max)	
Antenna cable (16 ft)	—1.9 db (max)	
Transmitting antenna gain	27 db	±0.5 db
Polarization loss (1 db axial ratio on transmit and receive)	0 db	±0.1 db
Pointing loss	0 db	0 —0.1 db
Path attenuation (385,000 km at carrier frequency)	—211.4 db	±0.4 db
DSS antenna gain (at spacecraft transmit frequency measured at DSS mixer)	51.0 db	±1.0 db
Effective noise temperature	—23	±1.2
Maser excess noise 100°K ±1 db		
Source temperature 100°K ±1.5 db		
Boltzmann's constant	228.6	
Noise bandwidth (3.0 mc)	—64.8	
Signal-to-noise ratio in 3.0 mc IF bandwidth	14.7 db	4.2 db —3.3 db
Threshold (FM)	10.5 db	1.5 db
Center margin	4.2 db	5.7 db —4.8 db

$$(S/N)_v = 19.1 + (14.7 - 3.3) = 30.5 \text{ db}$$

The conclusions reached here indicate that the S/N ratio of the video is adequate based on the theoretical model. Hardware development will be directed toward improving estimated parameter values and experimentally confirming this calculation.

b. Center Margins of other Calculations. Table 11 lists the center margins of each calculation made for the spacecraft telecommunication link.

The signal-to-noise ratio of the high and low rate scientific modes, when taken with their tolerances, allow the possibility of a negative transmission margin. These calculations, as in the primary TV calculation include a demand that the signal-to-noise ratio in the IF amplifier be +12 db or greater. Again, the desired results is that the individual data channels be capable of transmitting data with a suitably high S/N ratio. The ability of the individual data channels to transmit scientific data was calculated based on an S/N ratio of 10 db in the IF amplifier. For the analog data channels, the S/N ratio, due to the transmission link alone, varied between 40 and 60 db. For the digital and pulse counting channels, the S/N ratio varied between 35 and 50 db.

Table 11. Telecommunications decibel margins

Calculation	Center Margin, db	Tolerance, db
A. Earth to spacecraft		
Transit phase transponder mode	41.0	6.5 —17.5
Command mode carrier	34.3	8.1 —19.5
Subcarrier	17.5	6.5 —17.5
B. Spacecraft to Earth		
Transponder mode	16.2	6.6 —15.9
Transit phase recorded data playback carrier	61.1	6.2 —15.2
Subcarrier (each of seven)	18.8	7.4 —16.5
Midcourse correction data		
Carrier	33.4	7.6 —17.0
Subcarrier	20.6	6.1 —15.7
Transmit phase, real time engineering data		
Carrier	26.4	8.2 —17.6
Subcarrier	16.5	6.7 —16.3
High data rate scientific mode	4.2	5.7 —4.8
Low data rate scientific mode	4.2	5.7 —4.8
Television transmission secondary mode	4.2	7.1 —6.1

Accordingly, as in the case of television, the basic requirements are satisfied based upon the theoretical calculation. However, the hardware development experimentation will continue to be directed toward improving the S/N ratio in the IF amplifier.

c. Parameters used in the Center Margin Calculations. Table 12 lists parameters used in the calculations of the center margins that are not listed in the sample calculation.

3. Planar Array (High Gain) Antenna

The primary function of the planar array antenna is to concentrate the energy of the wide-band television channel in the direction of the Earth. It will be used for transmitting some telemetry data. It is located high on the central mast of the spacecraft.

Table 12. Decibel margins

Parameter	Value	Tolerance
DSS transmitting power (maximum)	40.0 dbw	0 —1.0 db
DSS transmitting antenna gain	50.0 db	±1.0 db
Spacecraft receiving antenna gain	2.0 db	±2.0 db
Spacecraft antenna pointing and polarization loss	0 db	0 —10 db
Spacecraft receiver cable loss (8 ft)	0.95 db (max)	
Spacecraft receiver duplexer and interconnecting cable loss	—1.5 db (max)	
Spacecraft receiver phase locked loop bandwidth	—20 db	±1 db
Spacecraft receiver noise figure	—13 db	±1 db
Phase locked loop threshold	8 db	±1 db
Spacecraft receiver bandwidth (FM)	—41.1 db	±0.5

This high-gain antenna is a slotted-waveguide planar array. Circular polarization is accomplished by utilizing crossed slots and simple slots in a manner that produces two orthogonal linearly polarized planar arrays in a given aperture. These two interlaced arrays are fed in time quadrature. Higher-order beams characteristic of large slot spacing are eliminated through the use of waveguide with wave-slowing corrugations which in turn allows reduced slot spacings. The gain will be 27 ± 0.5 db; the antenna has a 3 db beamwidth of approximately $6\frac{1}{2}$ deg and it is elliptically polarized with an axial ratio under 2 db.

The last reporting period covered development of the linear array aperture plates. The beam tilt reported then was about $7\frac{1}{2}$ deg. A first-order correction was made in the slot spacing of the linear arrays to bring the beam tilt closer to 6 deg. These linear arrays have been designated as 1c and 2c. Patterns of a modified 1c array are shown in Figure 54. Its beam tilt is 6.3 deg. This is compatible with the space coverage required by the antenna positioning mechanism.

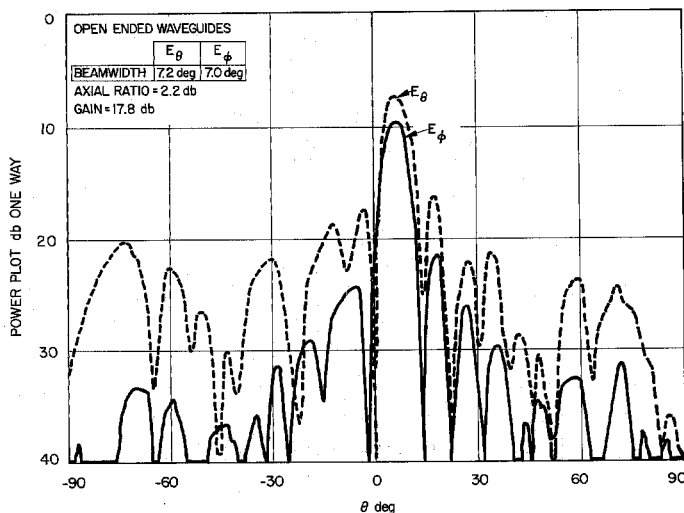


Figure 54. Antenna pattern of linear array 1c

Work on the feed line has progressed satisfactorily. The desired slot conductance values were obtained through adjustments of slot length and width, and the addition of inductive posts in the waveguide. The post-slot combination has worked out very well. It produces a resonant configuration and a sufficiently low conductance despite a slot location usually associated with conductances greater by an order of magnitude. The desired conductances and a very good match in the feed line have been achieved.

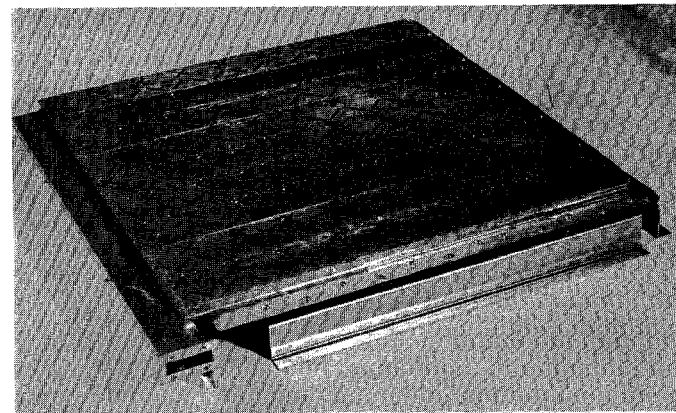
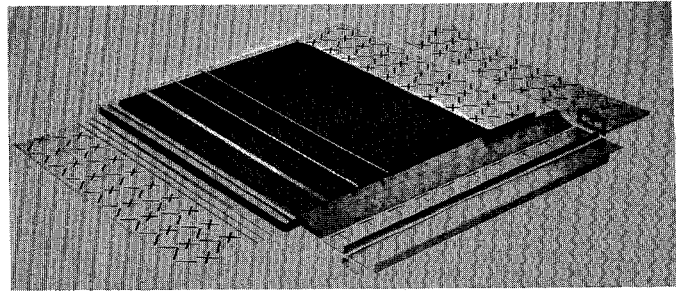


Figure 55. Electrical prototype planar array — disassembled (top) and assembled (bottom)

Effort is now being directed toward adapting the loop coupling transition from coaxial line to feed guide so that it will have a good match to the slot-post loaded feed line.

Fabrication of the electrical prototype (breadboard) planar array was completed. Its aperture plate is removable for easy modification if needed. The feed guide used the slot-post combination described above, and was waveguide-fed from one end. Provisions have been made to replace the waveguide terminal with a coax-to-waveguide transition. Figure 55 shows the array before and after assembly. The planar array consists of five type 1c and four type 2c linear arrays. Some of the tests on the electrical prototype antenna have been completed. The voltage standing wave ratio (VSWR) was under 1.3 over the required frequency band. Short loads have been made for the nine linear arrays for testing purposes only. Figure 56 shows VSWR as a function of frequency with and without loads. Rough measurements of feed-guide power split indicated that the amplitude was uniform to within ± 0.7 db.

Preliminary pattern and gain measurements have been made. The coordinate system for the patterns is shown in Figure 57. Patterns for $\phi = 0$ deg (linear array cut) and for $\phi = 90$ deg (feed cut) without loads are shown in

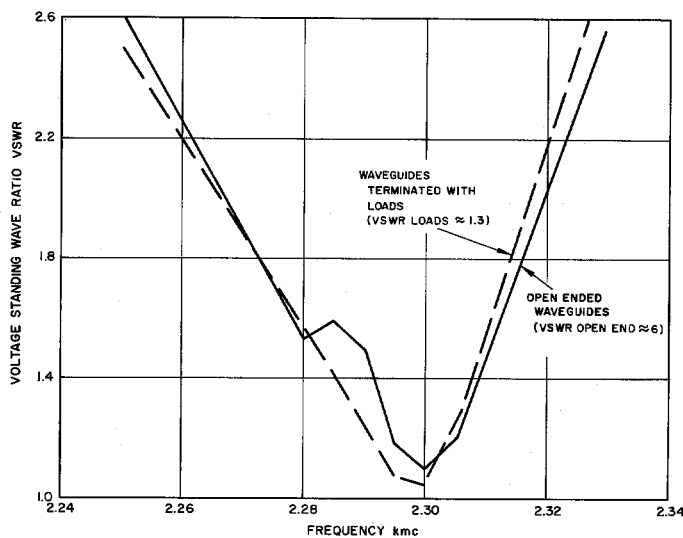


Figure 56. Voltage standing wave ratio of electrical prototype planar array (waveguide-fed)

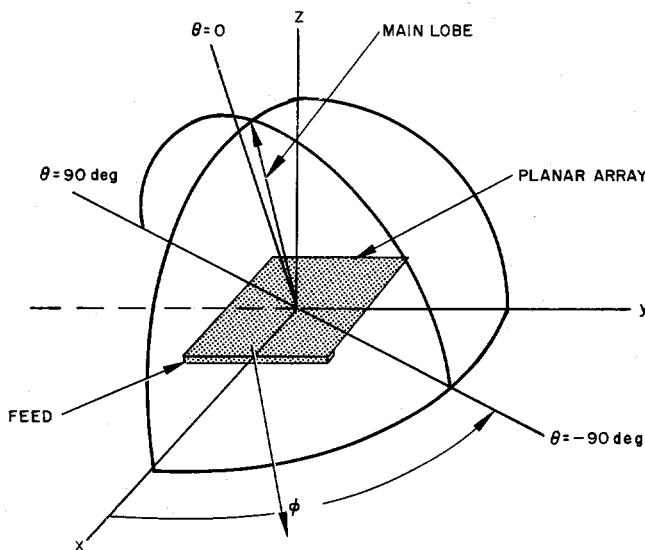
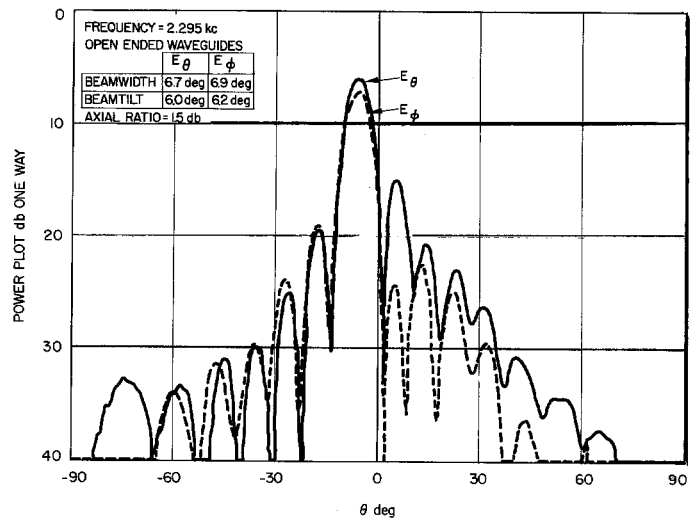


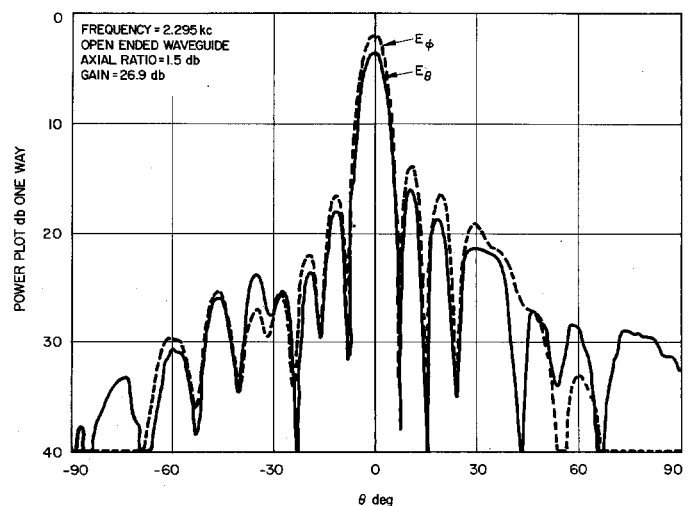
Figure 57. Coordinate system for planar array antenna patterns

Figure 58. They show an improvement over the linear array patterns. Axial ratio is 1.5 db and beamlift is 6.1 deg. On the basis of limited samples, the gain appears to be 27.0 db, implying an efficiency of 70%.

The polarization of the power leaking out the end of the array is aligned with the E_θ component for the $\phi = 0$ deg cut. This energy generates the high sidelobe in Figure 58a. Generally, near-in sidelobes are at least 12 db down; theoretically they should be -13.2 db. An



a) $\phi = 0$ deg cut



b) $\phi = 90$ deg cut

Figure 58. Antenna pattern of electrical prototype planar array

improvement can be effected in the E_θ components by using the short loads. In fact, this also decreases the axial ratio to 1.2 db, but it does not increase the gain. Present plans do not call for use of these loads in the final antenna design since sidelobe performance is not a factor.

Patterns will be taken at $\phi \approx \pm 30$ deg to determine whether or not the forward "end fire" lobes have been suppressed sufficiently. The gain indicates that they are satisfactorily low.

An ultrasonic welder mechanical prototype has been fabricated. This prototype as well as the previously com-

Table 13. Electrical prototype performance data

Item	Specifications	Measured
Gain	27.0 ± 0.5 db	27.0 db
Axial ratio	2 db maximum	1.5 db
Beam tilt	6 deg (no tolerance has been formally set)	6.1 deg

pleted resistance welded prototype are scheduled for vibration tests.

Progress summary. An electrical prototype planar array has been fabricated utilizing linear array designs 1c and 2c and it has been partially tested with a waveguide feed. Its performance, in this waveguide terminal model, is within specifications, as shown in Table 13.

The sidelobes are about as expected; although no specification was placed on sidelobe level.

Two different types of mechanical prototypes are ready for vibration tests.

4. Omnidirectional Antennas

The omnidirectional antenna system is designed to provide essentially spherical antenna pattern coverage about the *Surveyor* spacecraft. It is used in conjunction with the spacecraft command system, the doppler velocity tracking loop, the telemetry and television link, and the transmitter used for initial acquisition by the tracking station.

The omnidirectional antenna system consists of two circularly polarized low-gain antennas oriented with respect to each other and to the spacecraft. It is believed that this orientation will give the best spherical coverage and provide adequate operating characteristics in case the spacecraft tumbles.

Each of the two antennas is designed to provide broad pattern coverage (approximately 270 deg) in all directions except for a small cone behind the antenna within which the main bulk of the spacecraft lies. This type of pattern shaping minimizes the interference effects of irregular reflecting surfaces on the spacecraft.

Two different versions of the modified turnstile antenna are being developed, and both have conically shaped reflectors of 90 deg included angle. The first type (Fig 59) has the radiating elements oriented parallel to the reflector. This design requires that the reflector be spaced approximately one-quarter wavelength behind the elements. Impedance data have been taken using a single pair of radiating elements of approximately one-

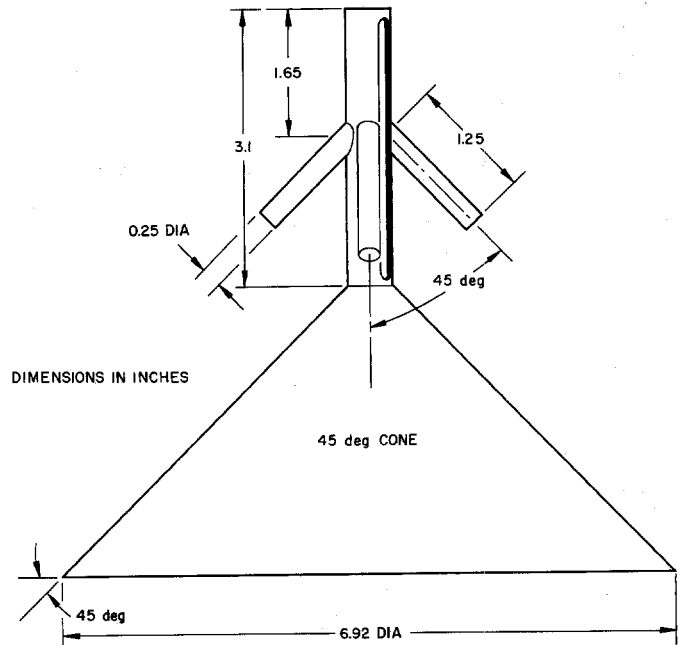


Figure 59. Omnidirectional antenna with elements parallel to reflector

half wave total length. These data were used to design the full-scale antenna presently being fabricated.

Circular polarization in this type of antenna is obtained by having one pair of elements less, and the other pair greater, than a half wavelength long. The impedances of the elements are chosen such that the currents lead by 45 deg in the short elements and lag by 45 deg in the long elements.

The second type (Fig 60) has radiating elements oriented at 90 deg to the conical reflector. This arrangement allows the reflector and element spacing to be minimized, resulting in a more compact, lighter weight antenna. It is also anticipated that this antenna type will have more satisfactory pattern coverage. A full-scale model using this arrangement has been constructed and patterns will be taken soon.

Some scaled work at X-band has been done on an antenna consisting of cavity-backed slots on a conical surface (Fig 61); it is believed that this design will give maximum control over the pattern although it is more difficult to feed. In the transit position, the solar panel now stands much higher above the spacecraft than it did on the quarter-scale model used for earlier pattern studies; this makes satisfactory coverage more difficult. Patterns taken with this antenna were very encouraging, as shown in Figure 62. A full-scale version of this antenna is now being fabricated.

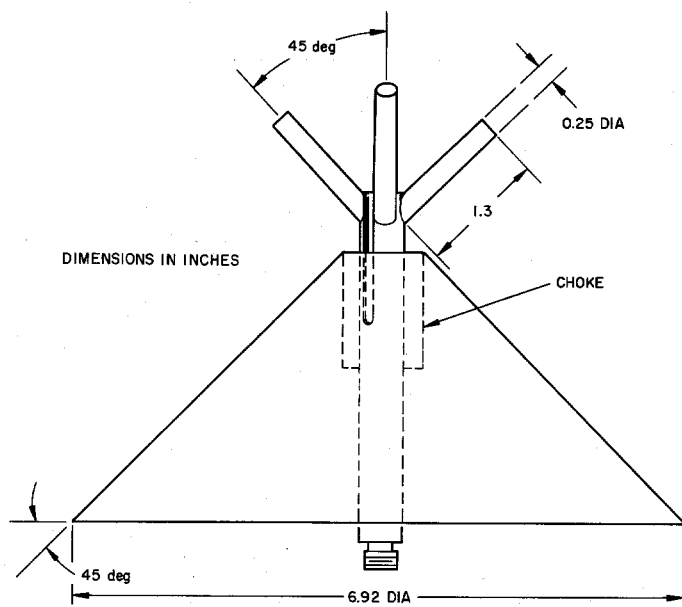


Figure 60. Omnidirectional antenna with elements 90 deg to reflector

Plans for modifying the antenna range are being put into effect. A new three-axis mount is on order, and contract will soon be let to modify the existing receiver site to accept this new mount. These new facilities will improve the speed and accuracy of obtaining omnidirectional pattern data with the full-scale model of the spacecraft. The model, MA-2, has been received, but a considerable number of changes in the spacecraft necessitates sending it back to the vendor for updating.

Progress summary. Full-scale versions of the two modified turnstile antenna types were designed. Type 2 has been built and type 1 is soon to be completed. A scaled version of the slotted cone antenna was built and tests proved the contention that very good control over the pattern could be obtained in this manner.

Antenna range modifications for MA-2 are being implemented. MA-2 has been built, but it is being updated while the range modifications are in progress.

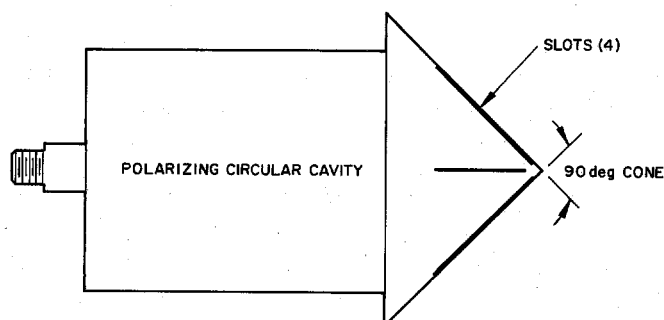


Figure 61. Slot-type omnidirectional antenna

Plans for the next report period consist of the following: measure antenna patterns of the two types of modified turnstile antennas, continue development of the slotted cone-cavity antenna, and continue to improve the broadband characteristics of the various full-scale antennas.

5. Transmitter

Two identical transmitters are included in each spacecraft in order to assure reliability through redundancy. Each transmitter consists of exciter circuitry which receives video signals and generates a modulated voltage at 19.125 mc. The modulated power is then amplified and multiplied to 100 mw at 2295 mc by means of solid-state circuitry. This output can be radiated directly or a traveling-wave tube amplifier may be switched into the path to provide 10 w of transmitter power. High or low power from either transmitter may be directed into any of the three antennas by means of RF latching switches.

The solid-state circuitry from the exciter up to 382.5mc utilizes lumped constant circuit elements and is packaged in four modules. Multiplication from 382.5 to 2295 mc is accomplished with distributed constant circuitry, and the circuitry is housed in a separate package which is mounted adjacent to the four companion modules.

a. Lumped parameter circuits. The lumped parameter circuits of the *Surveyor* transmitter have been divided

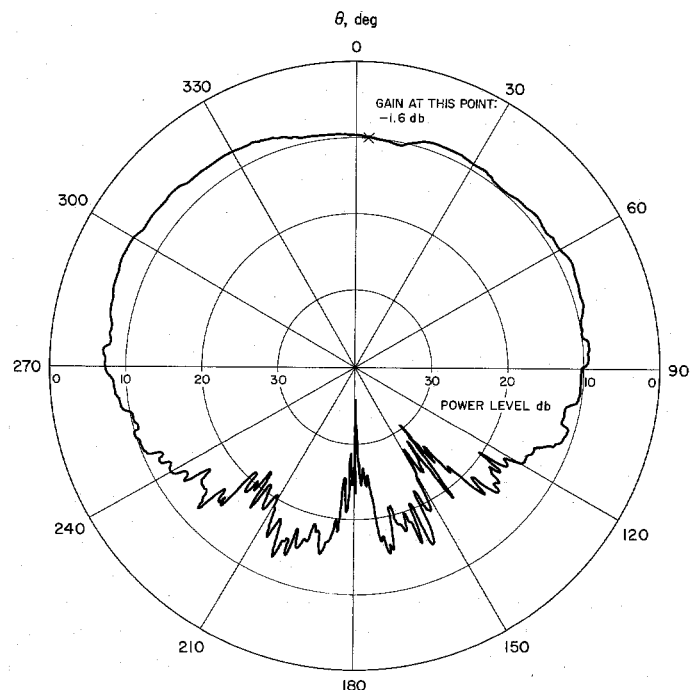


Figure 62. Slot-type omnidirectional antenna pattern

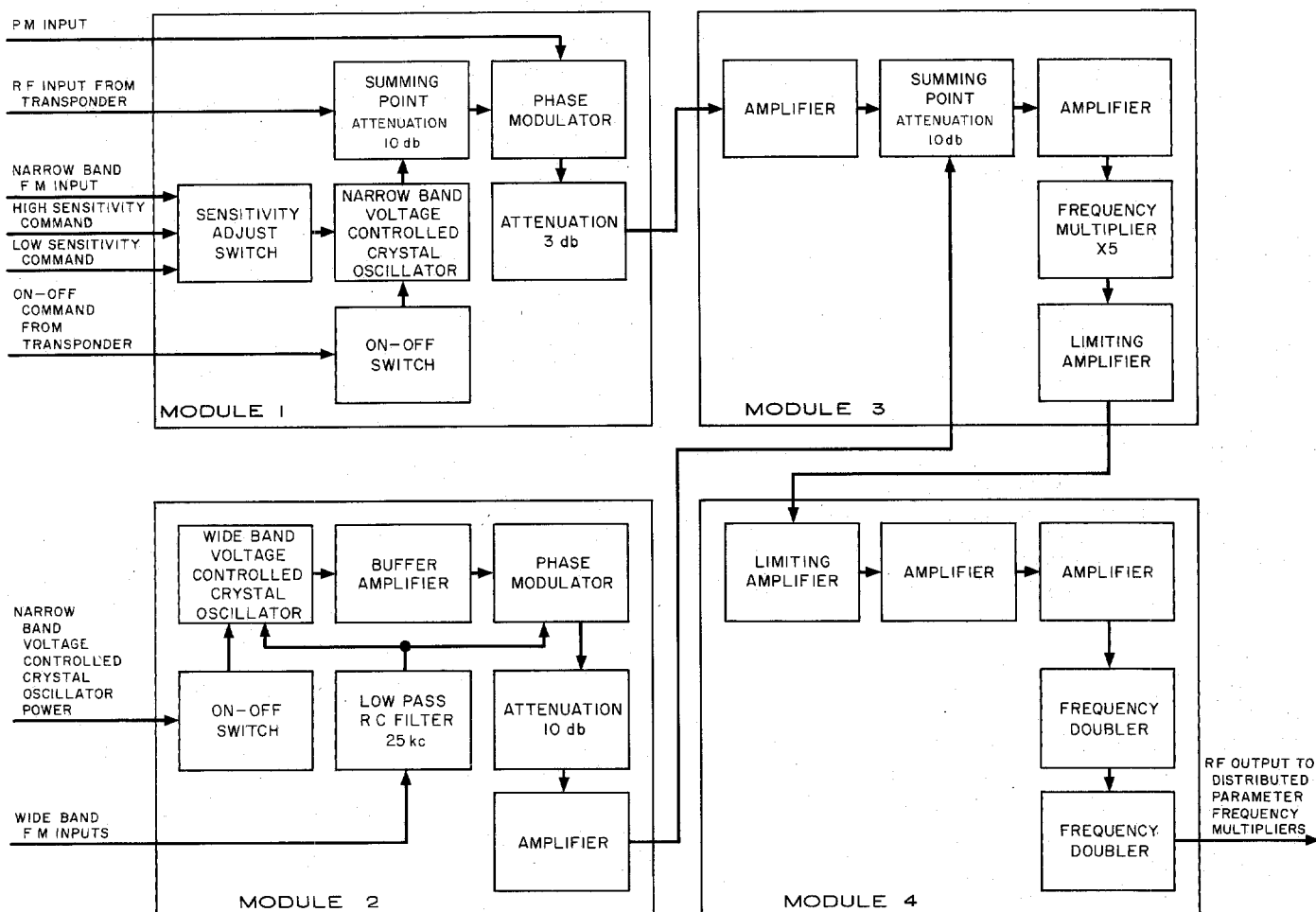


Figure 63. Transmitter lumped parameter circuits

into modules for final packaging. The block diagram of Figure 63 indicates the circuitry included in each module. The discussion below is in terms of these modules rather than by function as previously reported.

Module 1. Module 1 contains the narrow-band frequency modulator and the phase modulator. It is being temperature-compensated prior to final release.

Module 2. The wide-band frequency modulator is contained in Module 2. It is being redesigned to the block diagram in Figure 63. The original wide-band frequency modulator had a frequency response from 0.5 cps to 440 kc. Since there was no dc response, the modulation dynamic range had to be twice the television signal dc dynamic range. To stay within the communications bandwidth, the transmitter deviation would have to be reduced, and this would reduce the decibel margin in the communication link to an intolerable level. The solution to the problem is to dc restore the television signal

and provide a direct coupled modulator. This can be done with the modulator shown in Figure 63. The television video signal is passed through a low-pass RC filter and fed to a voltage-controlled crystal oscillator and a phase modulator. This provides a wide-band frequency modulated signal.

Module 3. Module 3 receives a signal from either Module 1 or Module 2. The signal is then amplified and multiplied in frequency by five. Module 3 has been released for final packaging. Performance data are shown in Figure 64.

Module 4. Module 4 contains a three-stage power amplifier and two varactor frequency doublers. This circuit has also been released for final packaging. Performance data are shown in Figure 65.

Two of the four lump-parameter modules have been released for final packaging. A third is undergoing tem-

perature compensation and should be released soon. The block diagram of the fourth module was changed recently in order to handle the television signal adequately. Circuit development is continuing on this particular module.

b. Distributed Constant Circuits.

×6 Multiplier. The addition of a resonant section of the multiplier output has been effective in reducing the sensitivity of the driver to load impedance change. The driver can now be switched among the traveling-wave tube, the diplexer, or the high-gain antenna line without degradation of performance.

The type of construction which will be used for the ×6 multiplier is shown in Figure 66. Formed aluminum sheet stock provides electrical ground planes and mechanical structure. This type of package, fitted with improved RF shorts, has been subjected to vibration, and preliminary results indicate that the phase shift caused by this stage is below the allotted level.

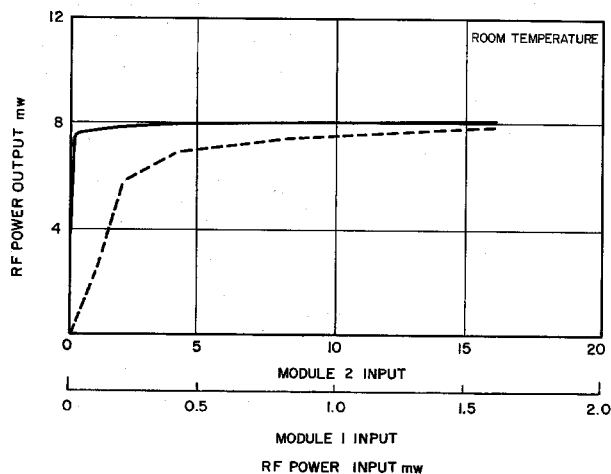
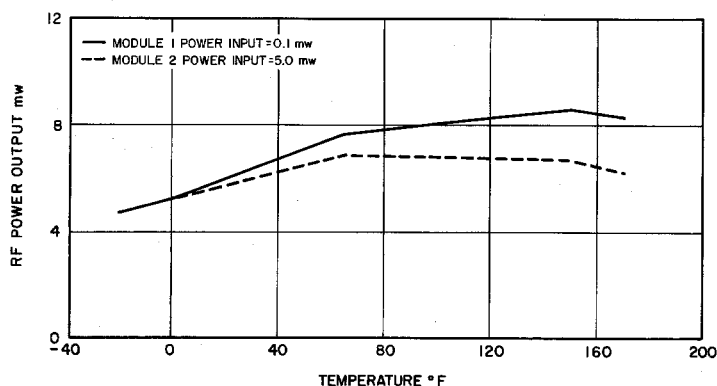


Figure 64. Performance data for Transmitter Module 3

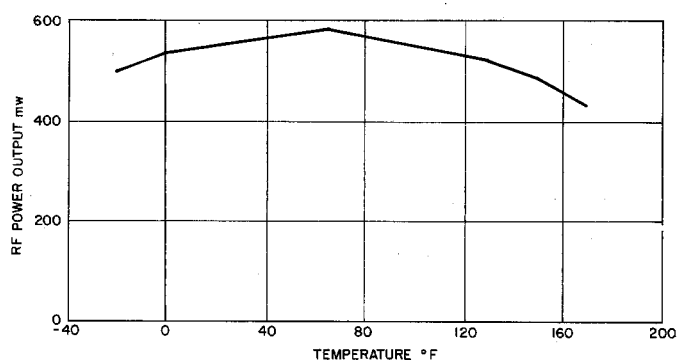
A complete chain in the final configuration is now being assembled for vibration and temperature testing with and without the TWT amplifier.

Traveling-wave tube. A breadboard tube has been operating with a solid-state driver for over 100 hr with 10 w output power at 2295 mc.

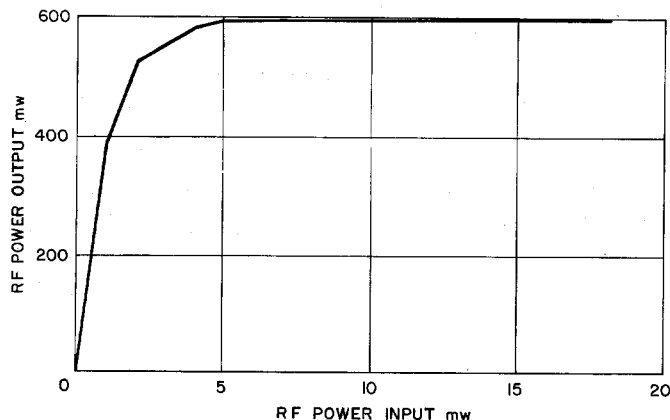
A sealed-off tube incorporating the low power (2-w) gun has been successfully built and operated. The first attempt to encapsulate the tube in the heat sink structure resulted in a cracked ceramic sleeve. The encapsulation technique has been improved and another tube is now in process. It is expected that vibration tests on this tube will be performed in February.

6. Command Receiver-Transponder Interconnections

The command link of the spacecraft telecommunications system contains two identical, independent com-



a) 5 mw power input



b) Room temperature

Figure 65. Performance data for Transmitter Module 4

mand receivers. Both receivers operate continuously throughout the life of the spacecraft to provide command capability. The outputs of both receivers are connected to the command decoder in which there is circuitry to provide continued operation of the command link in the event of a failure in one of the receivers.

Transponder interconnection circuits are associated with each receiver such that each receiver may be connected to a transmitter for transponder operation, wherein the transmitted signal is in phase coherence with the received signal.

The command receiver normally operates as a crystal-controlled double-conversion frequency modulation receiver. Commands are frequency modulated on a subcarrier which in turn phase modulates the main carrier of the Earth-to-spacecraft transmissions. The main discriminator in the receiver detects the subcarrier signal. The detected signal is fed to a subcarrier discriminator which reproduces the digitized command signals for the central command decoder.

The transponder interconnection circuitry may be activated by command signals. This causes the receiver to operate in a phase-lock mode in which the receiver local oscillator tracks the received signal. The receiver local oscillator signal which is phase-coherent with the received signal is connected to the transmitter exciter and transmitted back to Earth after the proper frequency multiplications have been applied to it.

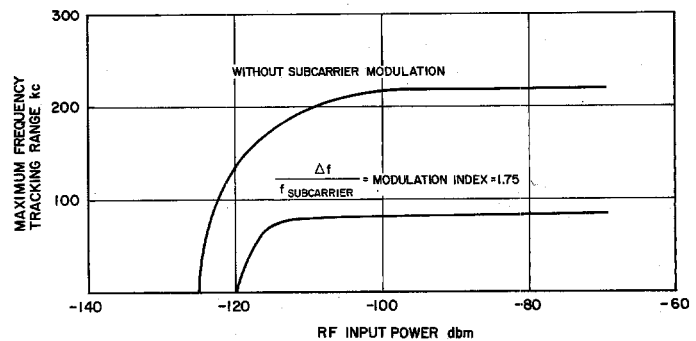


Figure 67. Transponder frequency tracking range

a. Breadboard receiver-transponder interconnections. The breadboard system, described in a previous bi-monthly progress summary, is undergoing quantitative tests and evaluation. Some of the test results are shown in Figures 67 through 69. Figure 67 shows the maximum frequency tracking range of the receiver in the phase-locked mode of operation. Figure 68 shows the FM-FM threshold, and Figure 69 shows the IF characteristics as a function of input power.

b. Crystal discriminator. The crystal discriminator was redesigned to improve linearity and reduce spurious responses on the S-curve. A circuit diagram of the crystal discriminator is shown in Figure 70. The discriminator consists of a tuned amplifier stage, two series resonant crystals followed by peak detectors, and a summing net-

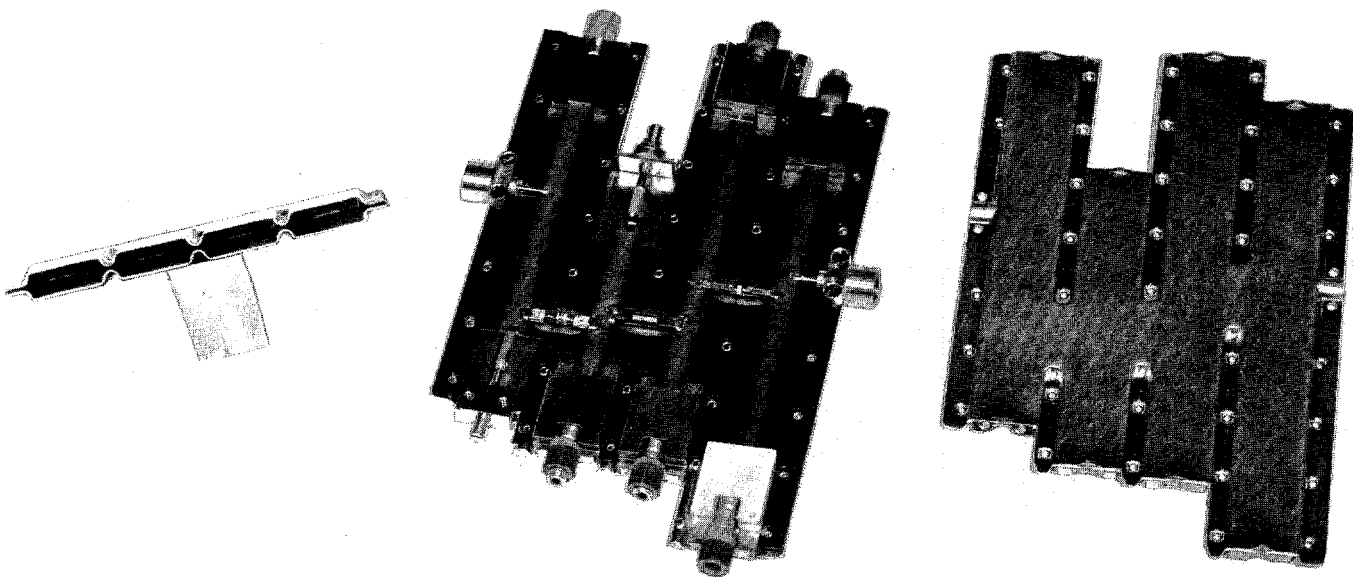


Figure 66. Multiplier

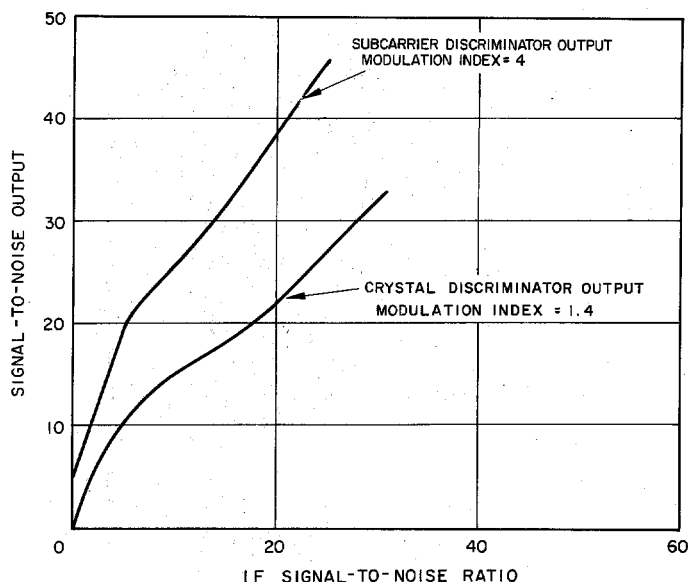


Figure 68. FM-FM threshold data

work. One of the crystals is resonant above the IF center frequency and is followed by a positive peak detector. The other crystal is resonant below the IF center frequency and is followed by a negative peak detector. The two detector outputs are added in the summing network. The discriminator has a 16 kc peak-to-peak frequency separation and a voltage sensitivity of 0.4 v/kc.

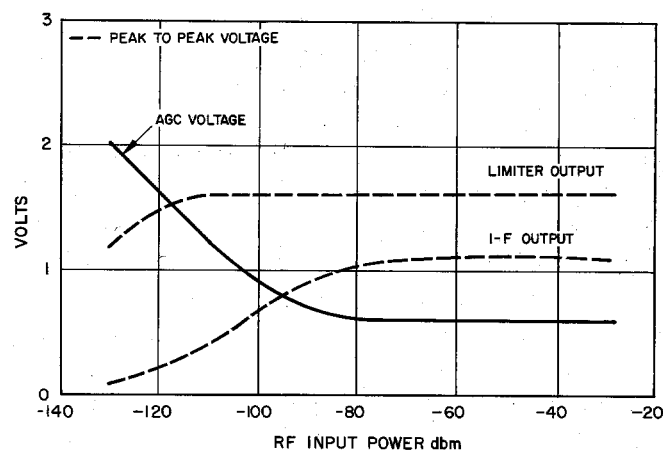


Figure 69. IF characteristics of breadboard receiver-transponder

c. Progress summary. The command receiver-transponder interconnections circuitry has been released to Product Engineering for layout and fabrication. Tests on the breadboard system are continuing and test data will be supplied to the product engineer for testing of the final packaged units. The breadboard system, packaged in a manner similar to the final version, is undergoing evaluation and environmental testing. The $\times 6$ multiplier for the solid-state local oscillator has been completed. The cavity-type multiplier is shown in Figure 71. The latest crystal discriminator circuit, which was recently developed, is being added to the breadboard system.

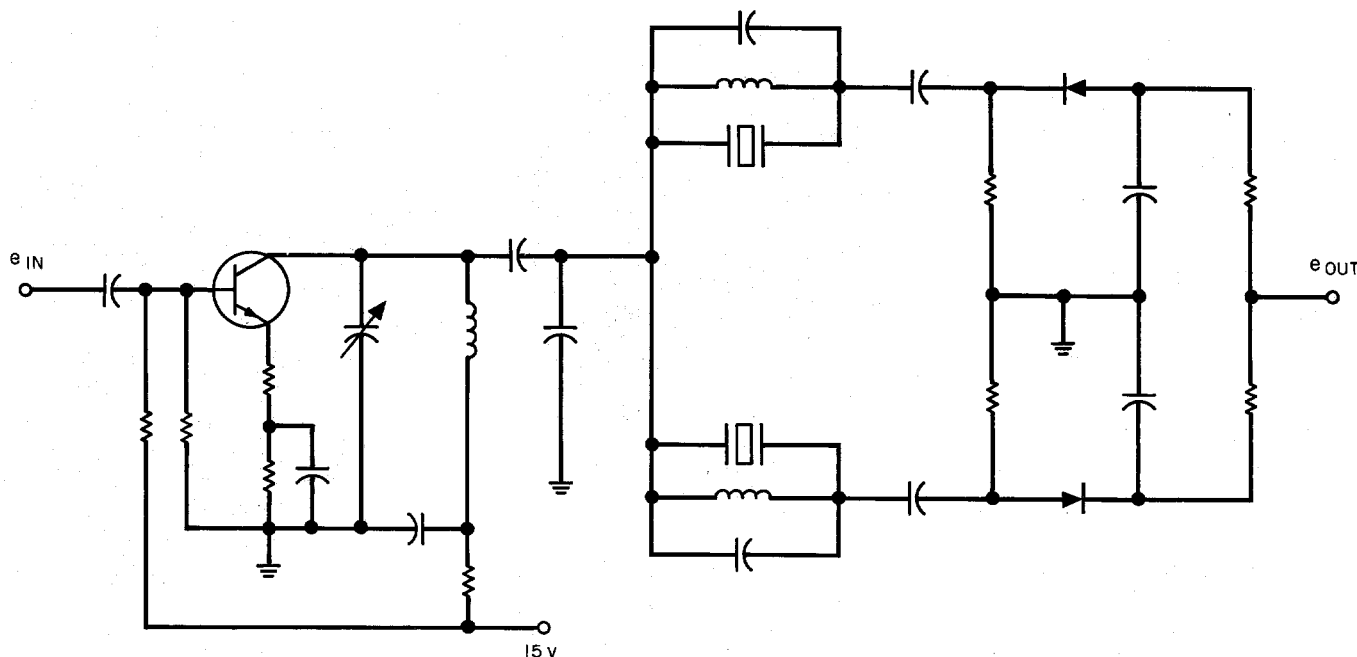


Figure 70. Crystal frequency discriminator

7. Signal Processing

The signal processing system processes the variety of signals from the scientific instruments and the many channels of engineering information into a form suitable for modulating the transmitter. The spacecraft-to-Earth telemetry link, of which the signal processing circuits are a part, uses an FM-FM and FM-PM subcarrier system. FM-FM is used after landing and FM-PM is used during transit. Wide-band FM is used for television transmission.

The signal processing system may be divided into three main categories: central signal processor, engineering signal processor, and scientific signal processing. The last category is composed of a multitude of signal processing circuits. The design goal has been to maintain the maximum degree of independence between experiments, therefore, individual processing circuits are designed for each particular instrument and are packaged with the instrument auxiliaries. The output of the individual scientific signal processing circuits are either subcarrier signals or signals suitable for analog-to-digital conversion.

Most of the engineering information from the spacecraft is transmitted in a pulse-code-modulation format. The engineering signal processor contains circuits for commutating the data into a single channel such that it may be converted to a digital form to maintain the required accuracy during transmission. Circuitry also is included to handle a few signals that require special processing.

The central signal processor is the common link in the processing system. The various subcarrier signals from the scientific signal processors and the engineering signal processor are combined and fed to the proper transmitter modulator. The analog-to-digital conversion process also is provided in the central signal processor. Digitized signals modulate a subcarrier oscillator and the resulting signal is combined with the other subcarriers in the central processor.

a. Engineering signal processor. A preliminary circuit design review has been held for the analog portions of the engineering signal processor. The circuits reviewed were the reject-enable circuit, the accelerometer processing circuit, the gyro speed countdown circuit, the constant current generator for the temperature measurement system, and the dc amplifier for the current measurements. Breadboards have been constructed and successfully tested.

The reject-enable circuit processes signals from the command decoders which indicate whether a command

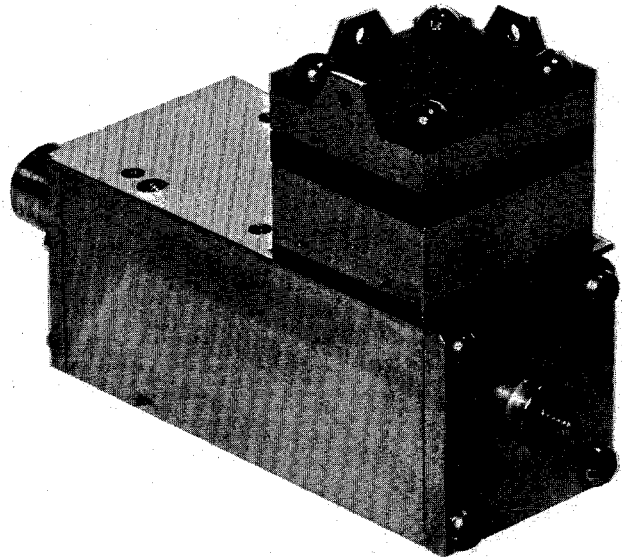


Figure 71. x6 multiplier-receiver local oscillator

is accepted or rejected. When a command is accepted and executed, an enable signal is received from the command decoder. This signal is processed by the reject-enable circuit and results in a negative 2.1 ± 0.4 -v, 21 μ sec pulse at the input of a 2.3 kc subcarrier oscillator. If a command is unacceptable, a reject signal is received and the reject-enable circuit delivers a positive 2.1 ± 0.4 -v, 21- μ sec pulse to the subcarrier oscillator. The pulses modulate the subcarrier oscillator, the output signal of which is fed to the central signal processor for combination with the rest of the telemetry subcarrier signals.

The accelerometer processing circuit consists of four subcarrier oscillators and a summing amplifier. The outputs from four accelerometers modulate four subcarrier oscillators (SCO). The SCO outputs are summed in an operational-type summing amplifier and fed to the central signal processor.

The 1600-cps output signals from the roll, pitch, and yaw gyros are processed by the gyro countdown circuit into a form suitable for transmission. The three-gyro speed signals are processed one at a time through the use of a four-position electronic stepping switch. The low-level 1600-cps signals are amplified and are divided by 32 in frequency to reduce the bandwidth necessary for transmission. The resulting 50-cps signal passes through a summing amplifier and then frequency modulates a subcarrier oscillator.

b. Engineering temperature measurements. Temperature measurements in *Surveyor* will be accomplished by

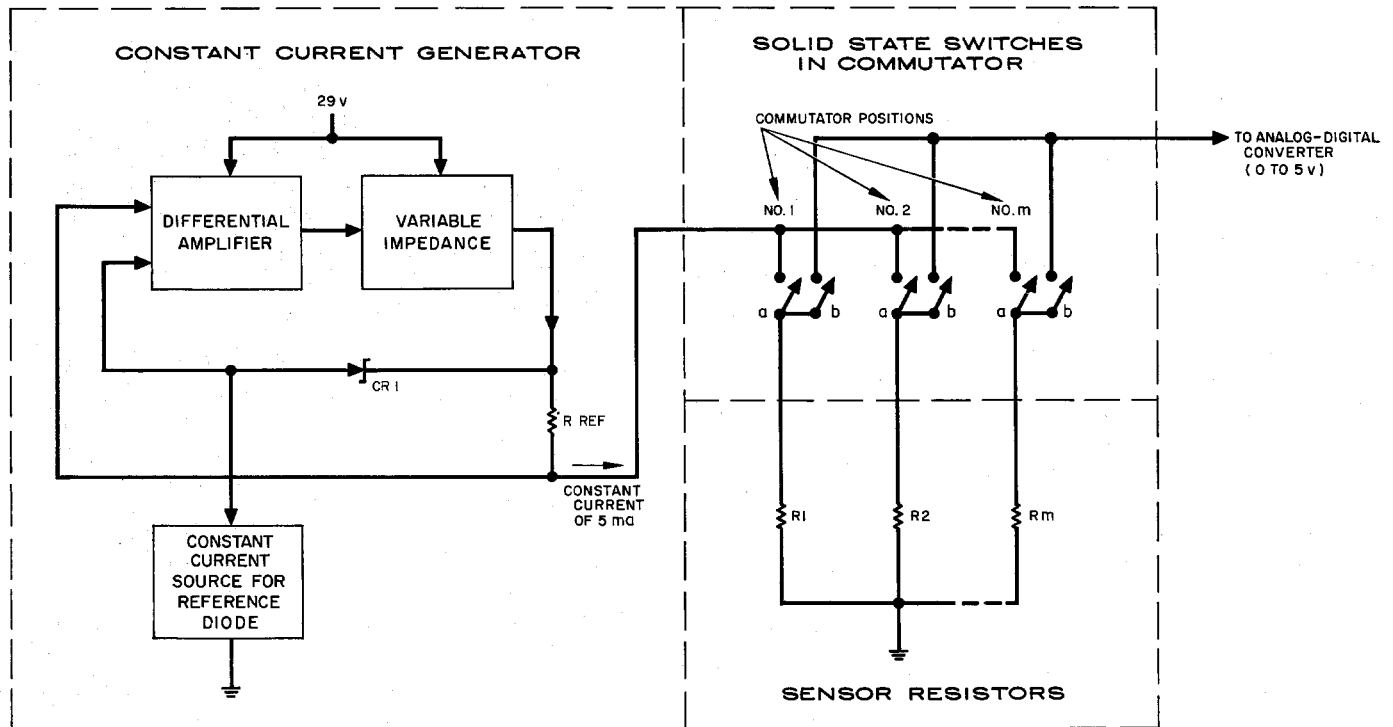


Figure 72. Basic temperature measuring system

resistive temperature measuring sensors. The resistive sensors are excited from a constant-current generator. The constant-current generator described in the previous bimonthly progress summary will be used only for a few "high" (1°C) accuracy measurements. In this current generator, the reference diode and reference resistor are located below ground potential and the sensor resistor common connections are maintained at a virtual dc ground. The current flowing through the selected sensor resistor must pass through the reference resistor except for a very small percentage that flows into the high impedance differential amplifier. Therefore, the current loading of the analog-to-digital converter does not increase the measurement error. The disadvantage of this system is that noise pickup on all the sensor resistors and connecting leads add together and will increase the error. This disadvantage is the reason a different type of current generator has been designed for the measurement of the majority of the temperatures.

The "basic" system that will be used for the majority of the measurements (accuracy requirements of 2°C or more) is shown in Figure 72. In this current generator, the reference diode and reference resistor are located above ground and the sensor resistor common connections are all returned to ground. A constant current is maintained through the reference resistor. Most of

this current will flow through the selected sensor resistor. However, any current loading due to the analog-to-digital converter will change the current through the sensor resistor and cause an error. This percentage error is minimized by increasing the current. A constant current of 5.0 ma has been tentatively specified.

The amount of error due to noise pickup on the long wires leading to the resistive sensors is reduced in both systems by a capacitor located at the output of the constant current generators. The commutator closes switches (a) and (b) for a time length of two words. However, the closing of the analog-to-digital converter selection switch is delayed in time, with respect to the closing of switches (a) and (b), by a time length of one word and then remains closed for one word. Therefore, the output of the constant current generator must reach 99.9% or greater of the final value within one word time. (The maximum bit rate during the temperature measurements is 550 bps or 50 words/sec. The time for one word is therefore 0.02 sec.) The maximum allowable capacitance at the output of the current generator is determined by the required response time.

c. Engineering signal commutation. Engineering signals to be transmitted to the ground via the telemeter system are sampled by commutators. Both analog and

digital signals are sampled and put in the proper word space of the commutation cycle or frame. Analog signals are sent to the analog-to-digital converter in the central signal processor to be converted to ten-digit binary numbers. Digital signals are grouped and sent directly to the output amplifier in the central signal processor.

Four different commutators are provided. Three of the commutators contain 100 words per frame and the fourth contains 50 words per frame. Any of the four commutators may be selected by radio command. Two "mode selection" flip-flops will be set upon command and remember which commutator has been selected. Engineering signals may appear in several of the commutators as well as in several words of a given commutation cycle.

The commutators consist of a flip-flop counter, selection gates, and electronic switches. The counter is made up of eight flip-flops and identifies each of the 100 (or 50) words of a frame. The count signal is obtained from the digit counter in the central processor. Each analog signal is sampled by an electronic switch. Selection gates from the counter turn on the electronic switches at the proper word times of each frame. The outputs of the switches are tied together and fed to the central processor where the analog voltages are digitized and transmitted at the proper time in the sequence.

On/Off signals are grouped into words with ten digits per word. Digit times from the central processor select the signals and place them in the proper position in the word. The word counter in the engineering data processor selects each group of ten signals at the assigned word times of the frame and feeds the data serially to the central processor. The frame sync pattern and a signal indicating that digital or sync data are being sent are also transferred to the central processor. The output of the Analog-to-digital converter will then be ignored by the output amplifier in the central processor and the digital information will be transmitted. The sync pattern is generated in the central processor and incorporated during the frame sync words.

d. Life tests — analog-to-digital oscillator. Simulated life testing of the crystal oscillator circuit for the analog-to-digital (A/D) converter timing circuits has entered its sixth week. In an effort to simulate sterilization conditions, the circuit was operated at $+125^{\circ}\text{C}$ for 4 hr. During this period the frequency was out of range, but upon cooling to room temperature, the frequency stabilized at the precise nominal frequency. This gives a preliminary indication that sterilization baking will not affect the NT cut crystal.

Noise tests were made on the oscillator circuit which showed that it was insensitive to power supply noise up to 1.0 v in amplitude at 25% duty cycle at frequencies near the oscillator frequency. Square wave noise at frequencies within 5 kc caused "lock-on" on the noise frequency at 0.75 v amplitude. Noise outside the 5 kc range had no noticeable effect on frequency at any amplitude. Capacitive decoupling of the oscillator from its supply voltages will be considered, if necessary, to ensure acceptable noise tolerance in the system.

e. Progress summary. Preliminary circuit releases of the central signal processor, described in previous reports, have been provided to the product engineer so that packaging layouts may be started. Final tests have been completed on the A/D breadboard in conjunction with commutator circuitry in the engineering signal processor. A complete breadboard of an A/D converter has been supplied to the subsystem integration laboratory for tests with other portions of the telecommunication subsystem. Final design of the A/D conversion circuitry is complete. The final design of the A/D timing circuit should be completed in March.

The engineering signal processor has been reviewed by the Design Review Board and requires minor changes before it may be released.

Circuit development is continuing on signal processing for the scientific instruments and should be completed within the next report period unless requirements are changed. Preliminary circuits of the acoustic velocity signal processing, previously described, have been released. Problem areas exist in the magnetic susceptibility and thermal measurements in the geophysical experiment. Requirements in these areas are not sufficiently defined to be certain that circuit development is proceeding along the proper approach.

The subcarrier oscillator specifications have been issued and proposals from several manufacturers are being evaluated. A contract should be issued soon. The specifications for the accelerometer amplifiers for the hardness experiment also have been released to Product Engineering and procurement of models is proceeding.

8. Command Decoding

The command decoding system of the *Surveyor* spacecraft accepts Earth-transmitted command messages from the spacecraft receivers, generates sync and timing signals from each command, checks each command for correct address and command complements, and provides output signals to command the addressed subsystems. The system has the capability to handle 960 ordinary

commands which control on-off operations. In addition, the system has the capability to handle a fill-in command and magnitude commands which control time-interval operations. The message format and the synchronizing circuits have been designed so that the system is synchronized with incoming messages after the reception of only one command.

The command decoding system consists of two redundant central command decoders (CCD), a receiver-decoder selector (RDS), and up to 30 subsystem command decoders (SD). The latter are associated with the various engineering and scientific instrument subsystems of the spacecraft. The central decoder unit (CDU) contains the RDS, both CCDs, and four of the SDs. The latter supply command pulses to the basic bus subsystems except the flight control subsystem, which has its own SD.

The RDS turns on an electrical conversion unit in one of the CCDs, thus applying several supply voltages to that CCD, when a message starts to come out of either or both receivers. The output of one of the two receivers is selected for decoding processing by a diode gate at the input of this central decoder. The selected receiver and the selected central decoder are determined by the states of two flip-flops in the RDS. These flip-flops form a two-stage binary counter having four possible states, each corresponding to one combination of one receiver and one central decoder. The counter is indexed to its next state every time messages stop coming out of either receiver. Thus, if one receiver and/or one central command decoder fails to operate normally, a different combination can be selected by interrupting the transmission of messages from Earth.

The major processing of the command messages occurs in the central command decoder. In order to synchronize the entire operation of the command decoding system, it derives sync and timing information from the incoming commands. Each ordinary command consists of a five-bit address, a five-bit address complement, a five-bit command, a five-bit command complement, and sync information. The CCD compares the address and command bits to their respective complements. Failure to check results in message rejection and initiation of a reject signal transmitted to Earth.

The address bits select and turn on one of the subsystem command decoders. After a correct complement check of a command, the command bits are decoded in a diode decoding matrix in this decoder. Each command selects one of 32 outputs of the matrix. The output signal is a 20.8-ms, 5-ma pulse of current which is directed to the designated load in the addressed subsystem.

Several changes have been incorporated into each CCD. (1) The previous command reject circuit, which was designed to drive an SCO, has been eliminated. A standard amplifier is now used instead to supply the command reject signal to the engineering signal processor. (Provisions have been made in the latter to send command enable as well as command reject signals to telemetry both by means of an SCO and by means of digital commutators). (2) Logic has been incorporated into the CCD to prevent accidental execution of irreversible commands, such as those which ignite squibs. A two-command sequence will be required. (3) The command bit rate has been decreased from 50 to 48 bits per second, at the request of command data console engineers. The resultant 0.5 sec command duration is convenient for timekeeping purposes.

A problem has arisen in connection with the current output pulse of the subsystem command decoder (SD). The existing specifications on rise and fall time are considered too tight and cannot be met under all expected conditions with circuitry of desired simplicity. Redesign to meet the specification will require one or a combination of the following: more components; the use of precision components instead of standard components (including heavy paper capacitors in place of solid tantalum capacitors); higher power dissipation; generation of an additional supply voltage which must be distributed to each subsystem decoder. It is desirable to increase the maximum rise and fall times from 2.0 ms each to 5.0 and 20.0 ms respectively. These longer rise and fall times are not expected to affect Hughes Aircraft Company equipment, which does not depend on the leading or trailing edge times. The change may affect some scientific instrument designs, however. The tape recorder was an example of this; alternate tape recorder circuitry has been suggested which not only makes the recorder insensitive to rise and fall times but also saves parts and, hence, weight and cost. Similar situations may exist in other scientific payload subsystems. An Engineering Change Analysis has been submitted to request the change.

Progress summary. Final circuit design of the command decoding system is in progress; completion is scheduled for late February. Detailed circuit design criteria and component specifications are complete. A breadboard of a CCD and four SDs has been delivered to the Telecommunications Subsystem Design group.

Final component holder and assembly drawings have been initiated. The initial layouts were studied to determine a format for all *Surveyor* module drawings.

9. Electronic Packaging Concepts

a. Introduction. The electronic product engineering function includes:

(1) Product design of basic bus electronic and scientific auxiliaries hardware.

(2) The selection of components, circuit layout, and module and chassis design in conformance with established standards.

(3) Mechanical design effort to establish shock and vibration requirements for each unit, to establish structural design goals for all equipments, and to perform detailed analysis and structural evaluations as the design progresses.

(4) Thermal analysis for optimizing the thermal control required by all equipments and the establishment of thermal design standards.

(5) The establishment of test equipment requirements, fabrication of test equipment, and flight acceptance and type approval testing of spacecraft hardware.

b. Packaging concept.

Unit installation. The units in compartments A, B, and C will be bolted to a thermally controlled mounting plate within the compartments. Standardization of connectors, chassis, and other parts of units will exist where possible. The units will consist basically of a supporting framework, chassis, connectors, and special components not suitable for chassis mounting. The units will not have covers except where shielding is required.

The flight control electronics unit will be attached to a T-section frame which is part of the spacecraft structure.

The altitude marking radar is mounted on the exhaust end of the retrorocket nozzle. It consists of radar electronics rigidly mounted to a 15-in. diameter platform which is the rear surface of a 30-in.-diameter antenna dish.

The TV cameras are mounted to the spacecraft by a ring of spherical radius surrounding the camera body. Brackets clamping the rings will allow alignment of the cameras on the spacecraft.

Chassis construction. The modules mounted on chassis will be removable and disposable in case of failure. Components are bonded to the plate and board sides. The foam-sandwich type chassis consist of a thin aluminum plate and a thin epoxy glass etched-circuit board bonded to either side of a $\frac{1}{8}$ in. sheet of polyurethane foam. The chassis will be assembled by installing the board side

components, bonding this board assembly and the chassis plate to either side of the sheet foam, and installing the plate side components.

The heat transfer path within the chassis is from components to the aluminum chassis plate. Plate-side components are mounted directly on the aluminum plate and good thermal tie is provided by the component mounting device. Most components will be attached by epoxy fillets, but provision will exist for clips, brackets, studs, and bolts for components requiring special mounting. Most components dissipating a significant amount of heat will be located on the plate side of the chassis. Board-side components will be attached with epoxy fillets. Heat transfer between board and plate will occur principally through copper leads. Aluminum extrusion angle brackets are used to provide heat paths between chassis and the heat sink. In Compartments A, B, and C the heat sink is the compartment surface to which units are mounted.

Module construction. Components are interconnected within modules by soldering the leads to circuit lines on 0.011 in. thick, epoxy-glass circuit boards containing circuitry on one side only. The modules have a height limitation of 0.5 in. with lateral dimensions between 1.0 and 2.0 in. The height was chosen to allow cordwise mounting of the high usage components, i.e., glass jacket diodes and metal film resistors.

Transistors with leads exiting from one side and long components are mounted parallel to the etched-circuit boards. To minimize the weight of the module a "freeze coat" resin application of a foam encapsulant will be utilized rather than a solid encapsulant.

Leads exit the modules in groups, in a close-spaced pattern. The grouping of leads will minimize the number and size of cutouts in the metal plate and minimize layout restrictions on board-side components. Components will not be mounted over module leads to maintain access to the leads for module removal.

The module leads are soldered into plated-through holes in the chassis master etched-circuit board. A "soldering-sucking" procedure has been developed for readily unsoldering individual module leads. An alternate procedure employs drawing of the solder out of the joint by capillary action into a heater flux braid. The modules are adhesively bonded to the plate, possibly using a tape. Most chassis will contain both modules and individual components. For units such as the central command decoding, where practically all chassis components will be modules, there will probably not be any board-mounted components.

RF packaging concept. The lumped constant RF circuits will be packaged using a 10-mil aluminum "T-bar" similar in geometry to that employed in JPL designs. RF circuits occupying one side of the T will be decoupled with feed-through filters from the dc circuits occupying the other side of the T, which in turn will be decoupled from the dc input leads. The T-bar will be enclosed by a 10-mil aluminum housing; the entire unused volume will be filled with foam and the base of the T-bar soldered to the enclosing housing. All tuneable components will be accessible through the base of the T-bar to allow external tuning. Dry micro-balloons called "Eccospheres" which have the same dielectric constant as the polyurethane foam will be used to test the effect of foaming on the prototype units while maintaining the ability to change components. This packaging concept will satisfy the requirements of minimum weight while still meeting all spacecraft functional and environmental requirements.

c. Environmental design. The equipment is designed to meet levels of shock and vibration which are based on type approval test requirements. Levels are specified for chassis and components as well as for control items so that the ability to meet type approval test requirements may be established for subassemblies prior to testing the entire control item. The estimates are based on inputs to the rocket engine; certain transfer characteristics were assumed for the space frame structure and the unit structure. The vibration levels for assemblies such as those on the rear of the altitude marking radar dish, traveling-wave tubes, vidicon tubes, etc., must be determined on an individual basis.

The equipment is designed to meet type approval temperature requirements. The altitude marking radar electronics will be controlled to $10 \pm 10^\circ\text{F}$. Low-emissivity surfaces will be used on the exterior of the radar including the antenna but excluding the surface opposite the throat of the rocket to minimize heat loss during flight. The interior surfaces of the antenna dish will also be coated with a low emissivity material. Present analyses indicate that 10 w during a 3-hr period just before operation will be required to bring the temperature from -41 to $+10^\circ\text{F}$.

The flight control electronics unit will be controlled within a range of about 40 to 95°F . Heat is radiated from the front and bottom surfaces of the unit.

Within the TV cameras, heaters will be used around the face of the vidicon to control the average face temperature to $30^\circ\text{C} \pm 7^\circ\text{F}$ with a maximum temperature difference across the face of 1°F . Upward looking radiators will be used to remove heat from the TV cameras.

d. Discussion.

Foam sandwich chassis. Foam sandwich chassis successfully withstood the temperature extremes associated with sterilization and lunar night survival. Some bowing of the chassis was observed at the extreme temperatures. The bowing occurred because the coefficient of thermal expansion of the aluminum on one side of the foam was about twice that of the epoxy-glass board on the other side of the foam. Other sandwich materials with matching coefficients were considered but any deviation from the aluminum-glass combination resulted in increased weight for the same stiffness.

To reduce the bowing effect, a foam chassis was fabricated with a 10-mil aluminum plate on one side of the foam and a special circuit board on the other side. The circuit board had 6-mil single-sided epoxy-glass boards bonded to each side of a 10-mil aluminum plate. The holes in the chassis were plated through; this configuration substantially reduced bowing, weighed less, and was more rigid than the chassis with aluminum on only one side.

Vibration tests were performed on beam specimens of a foam sandwich chassis. The measured first-mode resonant frequency was within 6% of the calculated frequency.

Machining the aluminum chassis plates would be an expensive and time-consuming operation. A process is being developed to chemically mill the cutouts and outlines of the plates. Preliminary tests indicate a satisfactory process can be developed.

Modules. The thickness of the module etched circuit boards was reduced from 32 to 11 mils. This will reduce module weights by as much as 20%.

In normal module construction the component axes for cordwood mounting are normal to the plane of the chassis. For normal 1 in. thick chassis with $\frac{1}{2}$ in. thick modules, the maximum cordwood component length is 0.418 in. To permit the mounting of longer components in cordwood configuration a new module design was developed which provides component axes oriented parallel to the chassis. The components are interconnected to two etched circuit boards which are normal to the chassis. An etched circuit board forming the bottom of the module maintains uniform spacing between leads exiting the module and provides a third surface for circuitry.

The combination of the two types of module construction will efficiently accommodate any family of components adaptable for modularization.

e. Progress summary. Outline and mounting drawings have been released for all control items, wood mockups for M12 have been delivered for assembly on the spacecraft, and information for the thermal mock-ups has been completed.

The product design of the central power control system, the altitude marking radar, the transmitter, the command receiver and transponder interconnections, the TV cameras, and the flight control and ECU portions of the flight control electronics unit is progressing in layout and drafting. Primary effort to date has been placed on the establishment of component types and other details required to establish the general packaging configurations of the various control items.

Camera mounting and external alignment provisions have been agreed upon with VDI. Internal alignment requirements are being determined. Layout studies of the cameras are continuing.

Thermal analyses of various configurations of Camera 1 were made using the IBM 7090 computer. A configuration with upward-looking radiators gave acceptable results. The vidicon tube would overheat if the camera was operated for more than 1.6 hr at the solar angle resulting in highest camera temperatures. A nominal vidicon tube temperature of 40 rather than 30°C would allow operation under all hot conditions. The cameras could operate with the Sun as much as 85 deg from zenith if 10 w of heater power is provided for 1 hr prior to operation.

A thermal analysis of Camera 2 was prepared and is being computed.

10. Radars

a. Altitude marking radar. The purpose of the altitude marking radar is to indicate the proper *Surveyor* spacecraft to lunar surface slant range to fire the retro-rocket. The firing signal is generated at approximately 56 mi.

The altitude marking radar employs pulse modulation of transmitted microwave energy in which the transit time of the return signal is compared to a predetermined delay (corresponding to slant range). At the appropriate range, the presence of microwave energy in the delayed gate is detected and integrated. If a preset threshold is exceeded in two channels, a marking signal is generated as required by the *Surveyor* system.

During this report period, the radar breadboard was successfully integrated and operated in the roofhouse. Echos from the Santa Monica mountains were obtained at approximately a 30 db signal-to-noise ratio. Since this

radar recognizes only moving targets, a real target with the required characteristics is difficult to achieve. A synthetic target, synchronized from altitude marking radar, is injected into the receiver to evaluate the marking function; the breadboard system appears to be marking satisfactorily.

A final design review was held on the solid-state local oscillator. The final productized breadboard was displayed (Fig 73) and performance curves were presented. This local oscillator meets or exceeds all its performance requirements; furthermore, it appears to be possible to select its frequency versus temperature characteristics to compensate for the magnetron.

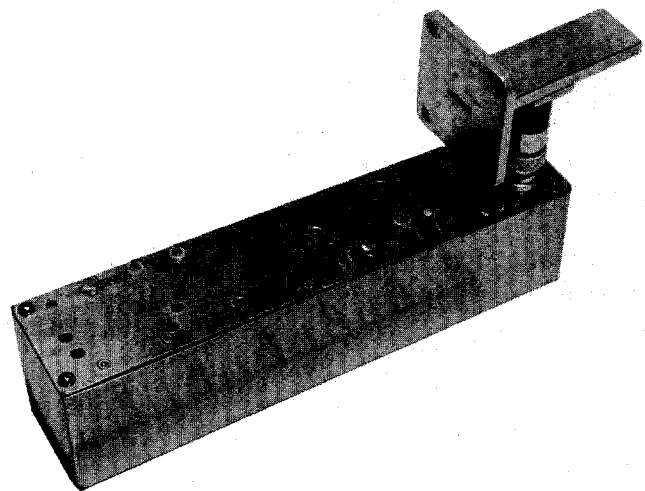


Figure 73. Solid-state local oscillator breadboard

A considerable amount of temperature testing of the RF assembly was accomplished during this period. Early tests indicated that temperature compensation of the magnetron was, on the average, very good but that frequency jumps occurred during temperature cycling. The anomalies were traced to two factors: the circulator and the magnetron tuning adjustment.

During testing of the first RF assembly, a sudden step in frequency was observed; to determine the cause, a Hughes circulator was substituted for the one originally supplied. It was found that with this circulator the RF assembly could operate into a 1.3 VSWR load with excellent characteristics; over the entire temperature range from -55 to +750°F, the frequency change was less than 1 mc with no anomalous changes.

The second magnetron was substituted in the test setup and again frequency discontinuities were observed during initial temperature runs. Subsequent examination of the magnetron revealed that the tuning mechanism exhibited stickiness such that temperature stresses could change the adjustment. When the temperature cycle was repeated without changing the frequency adjustment, this magnetron revealed a smooth frequency-temperature characteristic. The magnetrons under test have a wider tuning range than specified for the production configuration. It is felt that the redesign to reduce the tuning range will also eliminate the stickiness in the adjustment.

Except for the two problems discussed above, the RF assembly is deemed to be adequate for its mission.

A preliminary circuit release of the final synchronizer and video processing portion of the altitude marking radar was made to Product Engineering during this period. Minor modifications were incorporated to provide magnetron warmup prior to high voltage application, a main bang blanking pulse, integration of greater number of pulses, and test points.

To achieve the ultimate in magnetron reliability, it is desirable that the cathode be preheated before high voltage is applied. Therefore, the synchronizer has been modified so that the trigger to the modulator is interrupted until the enable signal is received from the flight

control system. The cathode is now preheated for the time from radar "on" to "enable."

For an off-vertical approach maneuver, the altitude marking radar will mark prematurely if strong echoes are received. The premature mark occurs because off-axis energy returns earlier than the desired axial return. For a 45-deg approach, for example, at a 60-mi slant range, 1 deg off-axis is a full mile closer than along the axis. To reduce the error associated with this effect, a peak AGC circuit is being added to the receiver. This circuit will reduce the receiver gain in the presence of strong targets to inhibit marking from energy returned from the skirts of the main beam. To implement the peak AGC, it is necessary that the main bang be removed from the receiver video. The removal will be accomplished with a main bang blanking gate supplied to the receiver from the synchronizer.

Because of the uncertainty in the reflectivity of the Moon, the altitude marking radar must be designed to mark reliably at a low signal-to-noise ratio; this requirement, of course, introduces a false alarm probability. Recent studies have indicated that there may be a more nearly optimum tradeoff between marking accuracy and reliability; that is, increasing the number of pulses integrated in the video processor increases the reliability of the mark signal at a sacrifice in accuracy. To permit productization to continue while the results are being

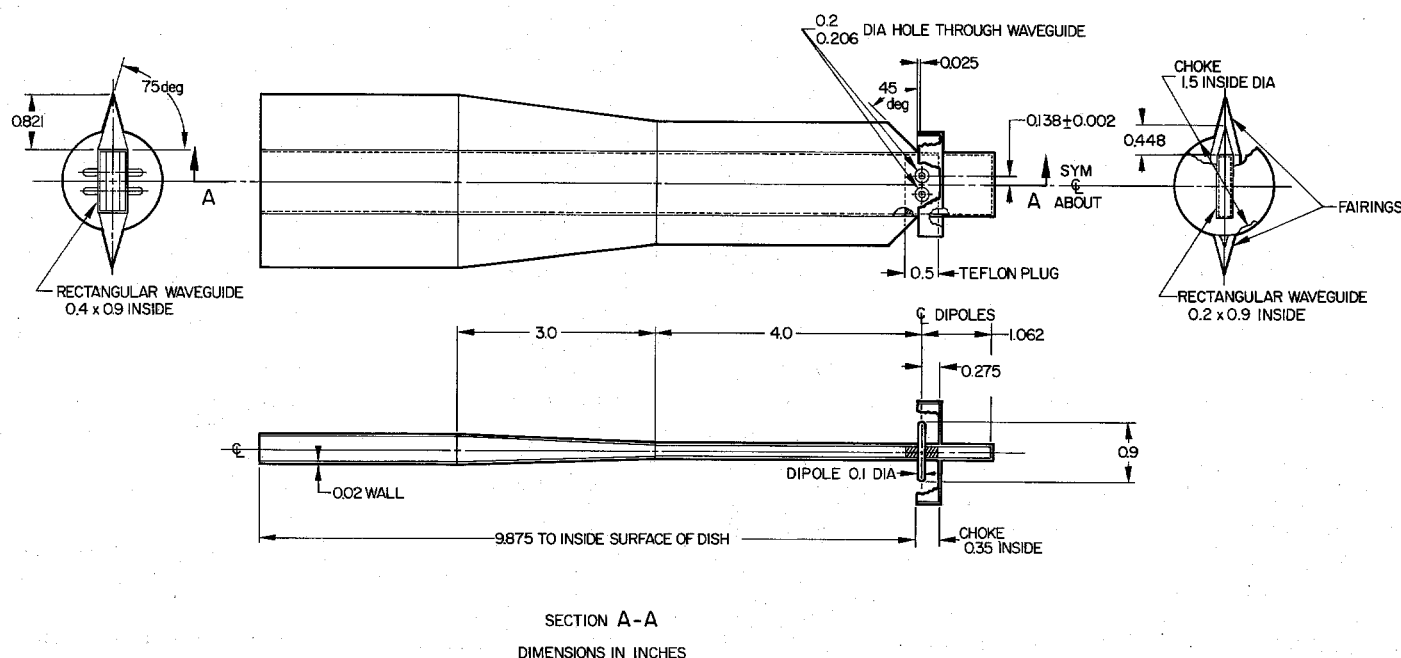
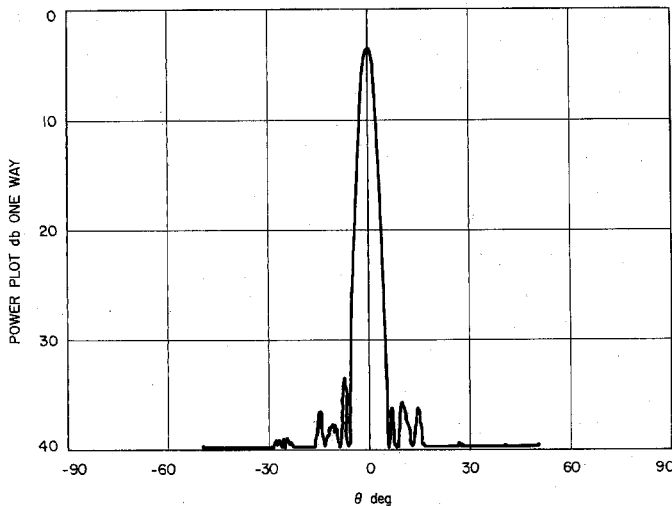
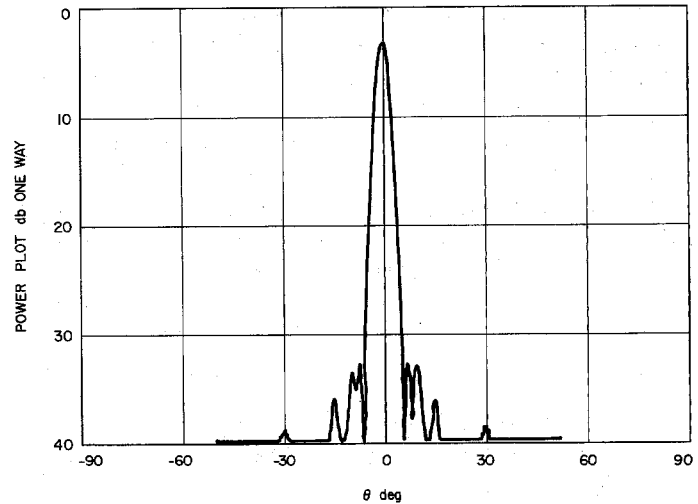


Figure 74. Altitude marking radar antenna feed



a) E-plane



b) H-plane

Figure 75. Altitude marking radar antenna pattern

evaluated, the video processor schematic has been modified to provide maximum integration of the number of pulses expected. In this way the designer will provide for maximum size of components, and the design will easily accommodate smaller components if a lesser number of pulses is finally selected.

When it was discovered that the regulated +29 v capacity of the spacecraft was being exceeded during the landing maneuver, it was decided that the altitude marking radar should be redesigned to operate from the unregulated +22 v bus. The decision was based on the facts that volume is available in the radar and that the weight penalty associated with the need for more regulated power would be minimized if it were operated on the unregulated bus.

The weight penalty is minimized because the altitude marking radar is not decelerated by the spacecraft engines nor is there need for extremely high efficiency in the electrical conversion unit. The electrical conversion unit for the altitude marking radar will consist of a switching regulator to provide a regulated dc voltage somewhat below minimum battery voltage and a dc-to-dc converter to provide the various regulated voltages needed by the radar. It will be designed as a lightweight low-efficiency unit; the efficiency will be the lowest practical which will not degrade reliability by excessive self-heating during system test and mission operation.

Antenna. In the last reporting period the problem of shadowing in the H-plane primary pattern was discussed. Extensive pattern studies have now been made on the

feed with fairings fitted along the narrow wall of the waveguide in an effort to eliminate this shadowing (Fig 74). The primary patterns showed little improvement, but the presence of the fairings made it difficult to obtain accurate data in close to the feed. However, the secondary patterns were improved by addition of the fairings. The sidelobe level in the H-plane was reduced approximately 1.5 db and the sidelobe structure was improved. With the fairings in place, it was possible to find a feed position where the E- and H-plane patterns were nearly symmetrical and a gain of 34.1 db was obtained. The highest first sidelobe obtained was down 29.2 db from the main lobe and the second and third sidelobes dropped off as expected for a properly illuminated antenna of this type (Fig 75).

A system analysis has been made based on these patterns. The analysis indicated that satisfactory system performance could be achieved with these antenna radiation characteristics. Hence, the design is considered satisfactory and a final Electrical Design Release is being written.

Some additional tests have been made to verify performance of a "productized" feed using 0.020 in. wall thickness instead of 0.050 in. It proved to be satisfactory from an electrical standpoint. Further testing will be done at the request of the Product Engineering Department when changes in the mechanical design to ease fabrication problems are suggested.

Summary. Secondary patterns in both E- and H-planes have been achieved which satisfy the system requirements. The final modified feed consists of double dipoles,

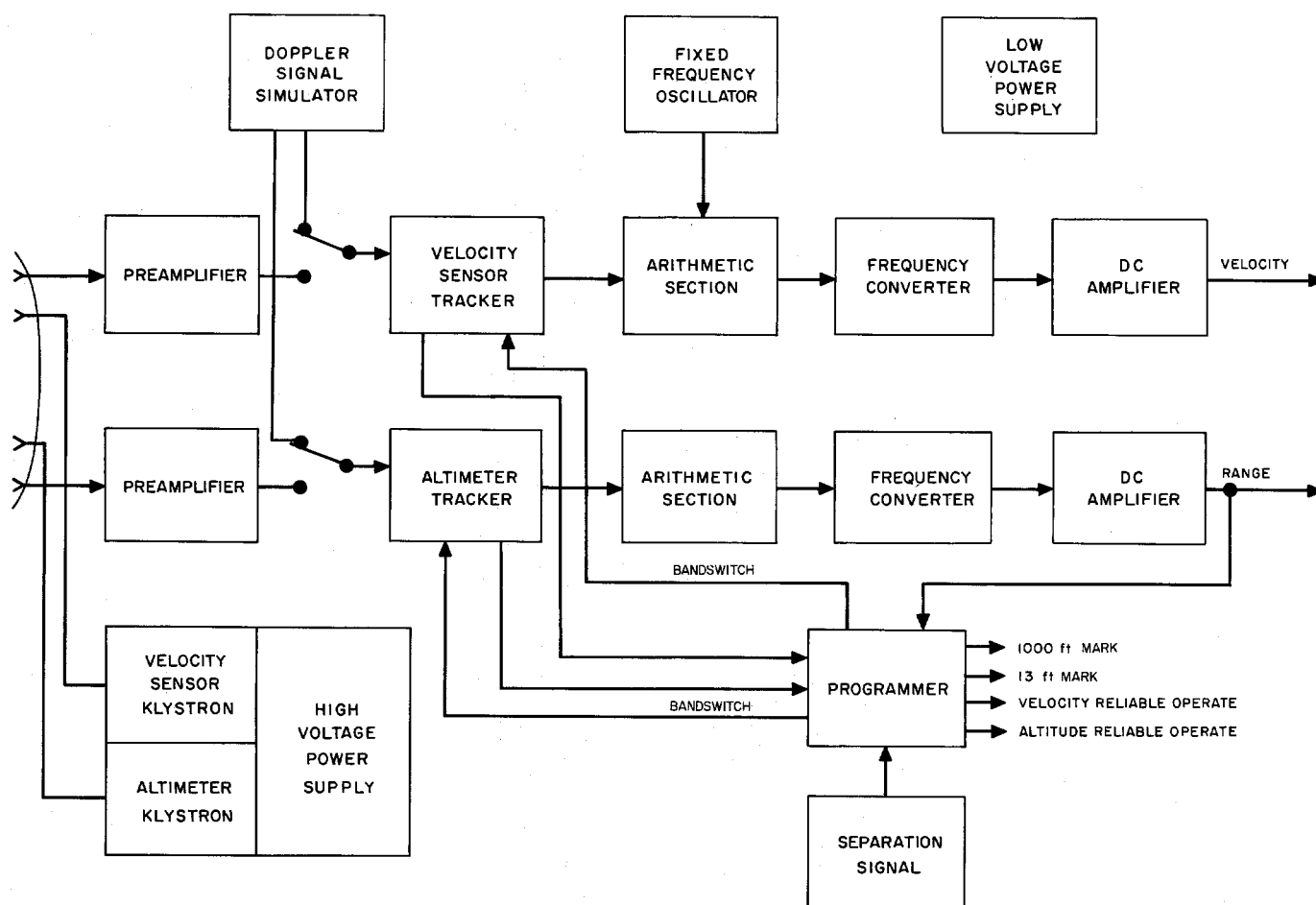


Figure 76. Radar altimeter and doppler velocity sensor system breadboard block diagram

a cup reflector, and fairings along the narrow wall of the feed waveguide. Development work has been essentially completed, but electrical testing will be continued as necessary to check out any changes which are desirable for ease of fabrication.

b. Radar altimeter and doppler velocity sensor. The purpose of the radar altimeter and doppler velocity sensor (RADVS), being developed by Ryan Electronics, is to provide spacecraft altitude, attitude, and velocity information to the spacecraft flight control system during the final portion of the lunar landing phase of the mission. The doppler velocity sensor utilizes three continuous-wave radar beams to obtain three orthogonal velocity components, referenced to the spacecraft roll axis. The radar altimeter utilizes a single frequency-modulated beam directed along the spacecraft roll axis to obtain slant range to the lunar surface.

The RADVS consists of four major units. The klystron/power supply/modulator (KPSM) includes the altimeter klystron transmitter tube and the doppler velocity sensor tube, the high-voltage power supply for the klystrons, and the FM modulator. The velocity sensor receiver-transmitter consists of an antenna which produces two of the three doppler velocity beams and which also contains microwave filters and isolators, microwave-to-audio detectors, and gain-controlled preamplifiers. The altitude-velocity sensor receiver-transmitter is similar to the velocity sensor receiver-transmitter except that its function is to produce the third velocity sensor beam and the altitude beam.

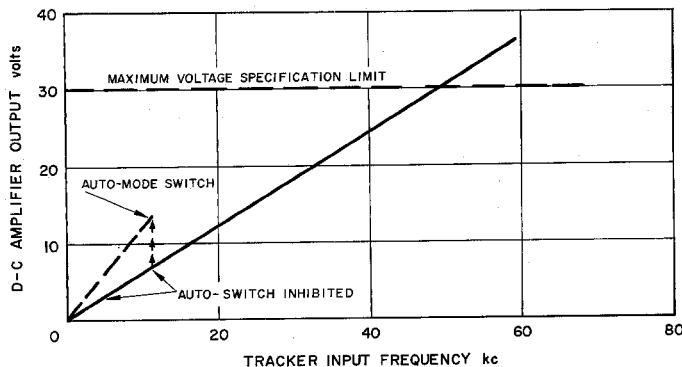
The fourth unit is the signal data converter (SDC). This unit performs the search and tracking functions for all four channels and processes the signals to produce the velocity, altitude, altitude mark, and reliable operation signals.

The breadboard units have been integrated into a breadboard system. Although the breadboard does not represent a complete RADVS system, it integrates all circuits necessary to prove system compatibility. Figure 76 shows the block diagram of the breadboard system. A space-duplexed antenna provides the altimeter beam and one velocity sensor beam. The recovered slant range and doppler signals are each processed through their separate preamplifier, tracker, arithmetic unit, frequency converters, and dc amplifiers.

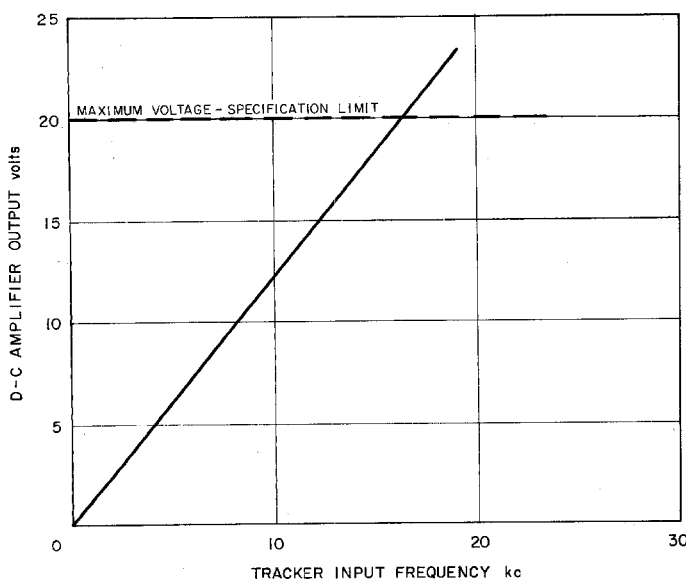
Preliminary altimeter breadboard system test results are illustrated in Figures 77 through 79. The FM deviation of the altimeter is switched from 4 mc (1.63 cps/ft) to 40 mc (16.3 cps/ft) as the range decreases through 1000

ft, slant range. System performance in the high-deviation mode is shown in Figure 77a. The low-deviation performance is illustrated in Figure 77b. These two curves, based on simulated targets, represent the range-proportional system output voltage as a function of the range-proportional frequency of the received signal. As shown in the curves, the system linearity is excellent and extends appreciably beyond the required range. Figure 78 illustrates receiver-transmitter performance with an actual target (hand-held variable range corner reflector). In this figure, the frequency is the dominant line in the FM spectrum as measured with a spectrum analyzer and corresponds to the frequencies in Figures 79 and 80. Also, data have been taken at 125 and 2200 (approximately) ft. The frequencies, for the high-deviation mode, at these ranges were 2.0 and 36.0 kc, respectively.

The system performance of the velocity sensor receiver, using a simulated signal, is illustrated in Figure 79. The maximum values of V_x , V_y , and V_z (roll-axis compo-



a) High-deviation mode, 16.3 cps/ft



b) Low-deviation mode, 1.63 cps/ft

Figure 77. Altimeter breadboard system performance

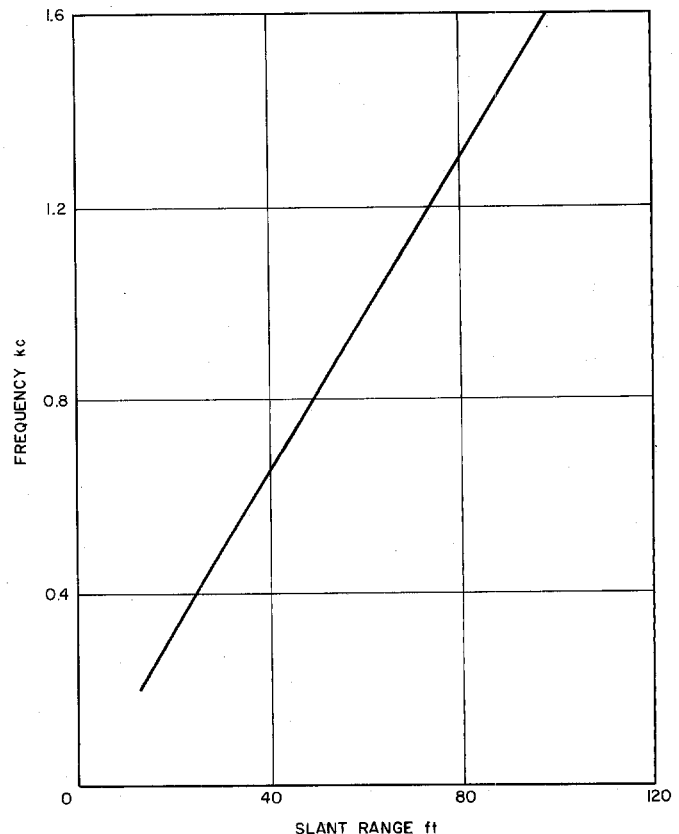


Figure 78. Receiver-transmitter performance

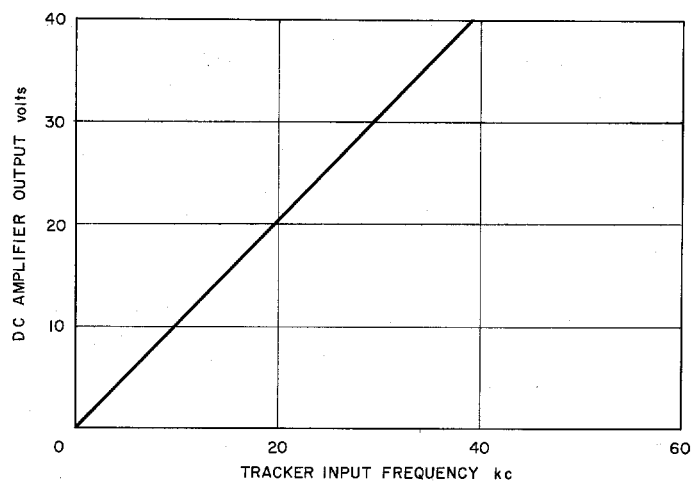


Figure 79. Velocity sensor receiver performance

ment) velocity components required to be measured by the RADVS are indicated on the curve.

The few remaining circuit design problems in the individual units which have been identified during breadboard testing are being solved and product design has progressed to the point that approximately 15 submodule designs (the total system contains 56 types of submodules) have been released for manufacturing of the development models. Approximately ten of these submodule

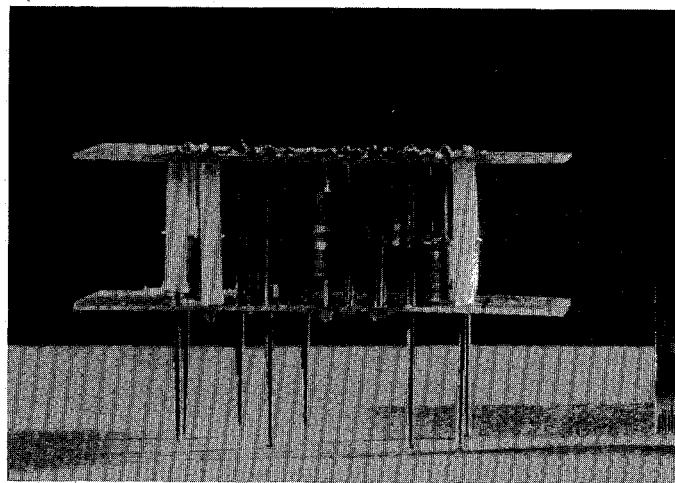


Figure 80. Typical submodule

types have been fabricated and tested. Figure 80 illustrates a typical submodule. The antennas for the development models have been completed and tested.

During this period, designs have been established to permit partial use of the radar data when one or two beams are in directions of zero velocity. In addition, a method has been established to provide lunar reflectivity telemetering information from the radar with minimum additional weight and complexity.

F. Electrical Power Supply

1. Introduction

The electrical power for the *Surveyor* spacecraft is supplied by an oriented (Sun-tracking) solar panel and two parallel-connected sealed, silver-zinc batteries. Lunar day power is supplied primarily by the solar panel, while the battery furnishes peak load capability and provides the prime source of power during the lunar night. The Phase I solar cell module development and type approval test program was completed on schedule by Heliotek Corporation, Sylmar, California. Completion of the Phase I program by Hoffman Electronics, El Monte, California, has been delayed approximately 1 mo. Fabrication of the first development model solar panel was completed in mid-December and testing has been initiated.

The first experimental model sealed silver-zinc cells have been fabricated and are undergoing initial electrical performance tests at the Electric Storage Battery Company, Raleigh, North Carolina.

2. Solar Panel

The Phase I solar cell module program to determine the ability of the solar cell module to survive the severe lunar environment was awarded to two vendors on November 9, 1961. Phase I was completed by Heliotek Corporation on January 9 with satisfactory results. The second vendor, Hoffman Electronics, has experienced difficulties in the conduct of the program and is expected to complete the

program in February. The Phase I program called for both vendors to test lots of 25 solar cell modules under air mass one sunlight conditions at Table Mountain, California. Sunlight intensity and spectral distribution measurements were made simultaneously with the electrical performance tests of the modules. Measurements of the sunlight intensity by the three separate pyrheliometers employed were within $\pm 1\%$. Spectral distribution data is still being analyzed but will require additional tests to firmly establish its accuracy.

A summary of the results obtained to date in terms of the conversion efficiency of the modules is presented in Table 14. The efficiency in the table is the average efficiency of the lot of 25 modules.

On the basis of the test results from Heliotek, the most degradation occurred as the result of the high-temperature vacuum test at 110°C for 7 days. This test accounted

Table 14. Module conversion efficiency

Vendor	Sunlight Efficiency, percent at 100 mw/cm ² and 25°C		
	Before type approval test program	After first two tests	After type approval test program
Heliotek	11.7	11.5	11.0
Hoffman	10.8	10.2	8.8

for approximately one-half of the total degradation experienced during the conduction of the test program.

Analytical studies were conducted during the Phase I program by Hughes Aircraft Company and the vendors utilizing the results of Phase I sunlight electrical performance tests and solar cell test data obtained from previous satellite programs. The purpose of these studies was to obtain an extrapolation factor for solar cell power output in going from Earth sunlight under an intensity of 100 mw/cm^2 to air mass zero sunlight (140 mw/cm^2). Results show that for the presently manufactured solar cells with a relatively high spectral response in the blue region of the spectrum, the extrapolation factor should be 1.21. This factor will be employed in the Phase II specification for the procurement of approximately 10,000 solar cell modules for the *Surveyor* program. Using this factor, it will be necessary for a module to be at least 10.9% efficient under Table Mountain sunlight after environmental testing in order to meet *Surveyor* requirements.

Additional experimental and analytical work is planned to further resolve the questionable areas in determination of the Table Mountain sunlight spectral distribution. In order to obtain the necessary experimental data, a sunlight optical and tracking assembly has been fabricated for use in conjunction with the Perkin-Elmer Model 112 U monochromator. Preliminary sunlight measurements will be performed on the Hughes Aircraft Company roofhouse to establish measurement and experimental techniques.

Fabrication of the first development model solar panel was completed December 15, 1961. The panel contains

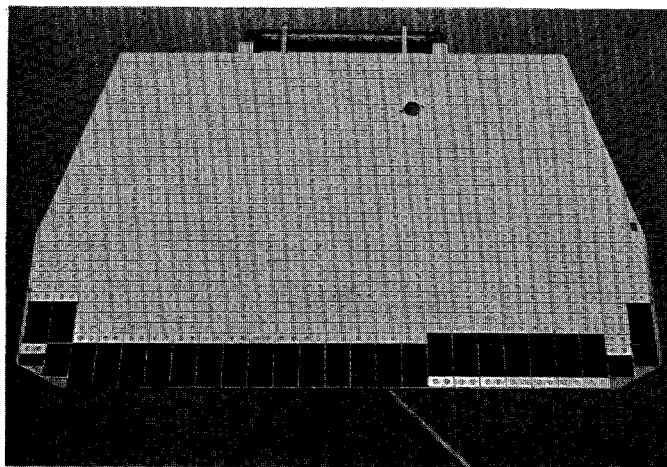


Figure 81. Development model solar panel front surface

106 series-parallel-connected solar cell modules assembled into a diode-isolated parallel subgroup equivalent electrically to approximately one-eighth of the full-scale solar panel. The solar cell modules utilized for this panel are approximately 1/2% less efficient than the flight quality modules. A photograph of the development panel front surface with some of the solar cells applied is shown in Figure 81. Spacing between the modules is 0.060 and 0.045 in. along the width and length of the module, respectively. Electrical interconnections and cabling is the same as that planned for use on the flight panels. The back surface of the panel contains the attachment brackets, flexible contour cable, terminal board, and secondary Sun sensor wiring. A view of the panel back-surface is shown in Figure 82.

Initial electrical performance tests of the panel have been conducted at Table Mountain. Test data corrected to 100 mw/cm^2 intensity and 35°C panel equilibrium temperature for one run are given below.

Maximum power	10.1 w
Short circuit current	193.5 ma
Open circuit voltage	73 v

These results, when corrected for efficiency and extrapolated to a full-size panel and air mass zero spectral

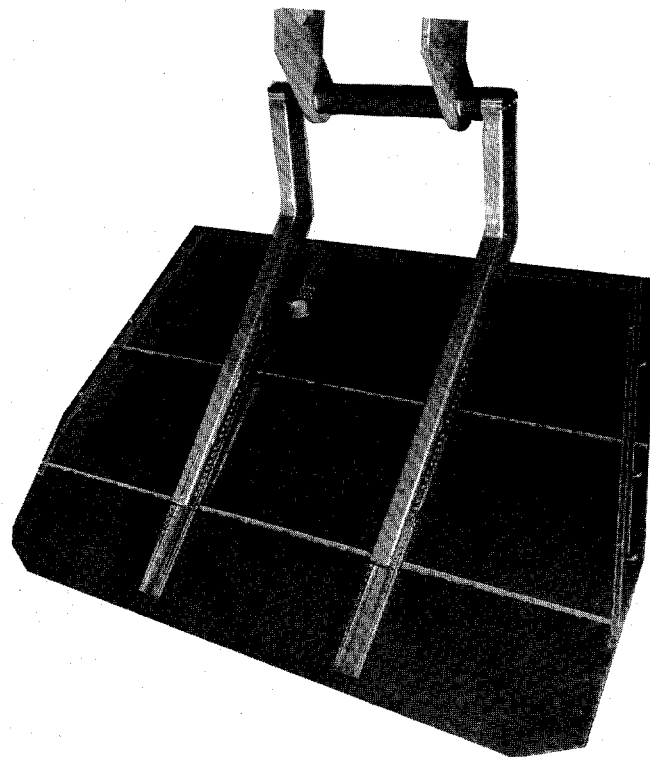


Figure 82. Development model solar panel back surface

conditions, demonstrated the capability of the solar panel to meet performance requirements.

Environmental tests to be performed on the development panel are humidity, vibration, shock, acceleration, and high and low temperature. It is planned to measure electrical performance before and after each test under a calibrated tungsten source to measure any degradation, and to conduct a final electrical test at Table Mountain.

A preliminary fabrication process specification has been prepared to establish detailed procedures for use in the fabrication of subsequent solar panels.

3. Solar Simulator

The solar simulator has been received from Optical Coating Laboratory, Inc. (OCLI) Santa Rosa, California. Preliminary performance measurements conducted at OCLI under Hughes supervision show that the simulator meets the specification requirements. Acceptance tests will be conducted by Hughes to verify that the solar simulator meets the procurement specification, and to correlate its radiation with Table Mountain solar radiation.

4. Battery

The silver-zinc battery program is progressing on schedule, both at the Electric Storage Battery Company (ESB) and at Hughes. The first technical coordination meeting was held on January 10, 1962 concerning scheduling, experimental program test details, and a firm definition of battery load requirements. Initial tests of experimental model cells by ESB indicate capability of the battery to meet requirements of the design specification.

Figure 83 presents a plot of current density per square inch versus cell voltage obtained from tests of the first experimental cell, which contained 175 in.² of plate area. The maximum current draw of 78.7 amp obtained from this cell corresponds to a minimum expected battery voltage of 18.0 v at 70°F. The plate area of the final experimental cells will be increased to 195 in.² which should enable the battery to meet the specification requirement of 85 amp at 18.0 v minimum and 70°F. A second critical design condition requiring a continuous 40 amp at 18 v and 30°F for lunar hole drill operation is also expected to be fulfilled with the larger plate area of the final experimental cells.

ESB procedures and techniques for producing sterile hardware, with the exception of a sporicidal epoxy, have

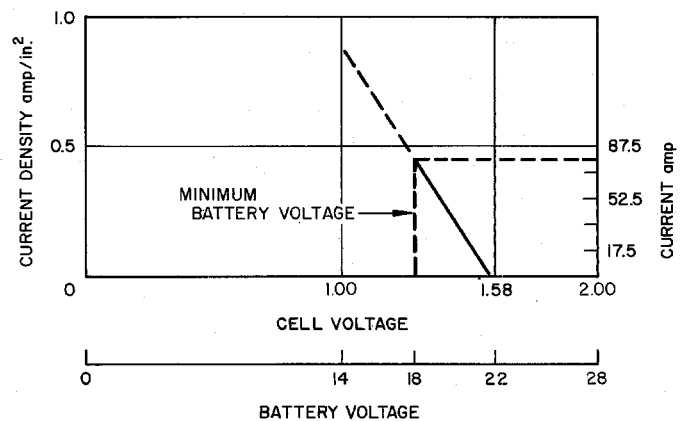


Plate area: 175 m²; temperature: 70° F

Figure 83. Experimental silver-zinc cell performance characteristics

been proved and documented for the JPL Ranger and Mariner battery. ESB is currently working with JPL to eliminate the contamination which apparently is characteristic of the epoxy utilized in construction of their battery cells.

Receipt of the first eight experimental batteries at Hughes is scheduled for March 1, 1962. Detailed test procedures are being prepared which will place primary emphasis on the electrical performance, leak-tightness, and heat dissipation characteristics of the battery.

5. Power Management System

The power management system is composed of a central power control unit and a variety of voltage converters, regulators, and power switches for the various instruments and subsystems.

The central power control unit provides conversion of power from the solar panel, main bus regulation, overload sensing, and battery charge control.

The optimum charge regulator circuit provides the most efficient conversion of solar panel power to battery charging current. The circuit tracks the maximum power point of the solar panel output characteristic which varies with the panel temperature. The panel is loaded in an optimum manner by the circuit to obtain the maximum possible transfer of power.

The battery output voltage, unregulated bus, is converted to the main regulated bus voltage by means of a boost regulator. The regulator consists of highly efficient switching-type circuitry which regulates the bus to the specified requirements.

A main overload trip circuit using tunnel diode current sensing removes all nonessential spacecraft loads when an overload condition is detected.

Battery charge logic controls the charging current to the two spacecraft batteries. Normally, the system operates in the automatic mode and batteries are charged until their voltages reach a predetermined level. A command mode is available where either battery may be commanded to be charged.

a. Boost regulator. Two functional deficiencies existed in the initial boost regulator design. The first resulted from a change in required maximum output capacity from 150 to 175 w. The second deficiency was a large loss at light loads although the efficiency was 90% at full load. Since much of the *Surveyor* operation during the lunar night is at low power, the amount of loss in the boost regulator is very significant.

The boost regulator has been redesigned in part and a new breadboard model is under evaluation. The major change in the design approach is as follows: The drives to all of the high current transistors in the original design were biased for full load operation; furthermore, the bias currents were designed for minimum gain devices in order to assure reliable operation. This biasing did not vary with load. Thus, at light load, excessive power was lost in the transistor drive circuitry. In the modified design, the drive for each high-current transistor is caused to vary with load current such that a constant forced current gain results and the light-load losses are considerably lowered.

The linear portion of the boost regulator has had relatively little change. There are only small losses in that area except during transient load changes in which the rate of change of load current exceeds the rate of change through the switching inductor.

b. Battery charge logic. The functions of the battery charge logic are as follows: (1) to maintain equal charge on the two spacecraft batteries so long as both are in good condition; (2) to provide independent charge cut-off for each battery when its terminal voltage reaches a predetermined level corresponding to full charge; (3) to allow command charging of either battery in an auxiliary charge mode; and (4) to allow command termination of charging when telemetered information indicates this need.

The maintainance of equal charge on the two batteries is automatic. Both batteries are diode-coupled to the unregulated bus resulting in the higher voltage battery delivering load current until the battery voltages

are such as to allow equal loading. Also, the charging is such as to supply all the available charge current to the battery with the lowest terminal voltage until the batteries adjust to allow equal charging. This automatic equalizing of battery charge can be interrupted in the event either battery is thought to be faulty as a result of interpretation of telemetered data.

In the auxiliary charge mode, either battery may be commanded into a charging mode. This command will cause all charging to cease in the battery *not* commanded to charge. During command charging, full voltage cutoff is not provided so discontinuance of charging must be done by command. This allows full charging of the batteries in spite of failure of the full charge cutoff; such failure could take the form of premature interruption of charging.

Referring to Figure 84, a detail of operation will be explained. Assume neither battery is fully charged due to current having been drawn from both batteries. The presence of a discharge current is sensed by the "battery discharge current sensing" circuitry. This compares the voltage at the anode of CR1 and the anodes of CR3 and CR5. This voltage must be less than a diode drop (approaching zero) in order for CR3 or CR5 to conduct. Therefore, this provides a sensitive indication of battery discharge. This voltage could also approach zero at no load when the drops across CR1, CR3, and CR5 are all zero; however, the boost regulator will always place a minimum load on the battery charge logic circuitry and prevent this.

The output of the sensing circuit, indicating a discharge has taken place, is fed to both charge bistables. These produce a gating signal which allows both blocking oscillators to run. The output of each blocking oscillator is a train of pulses of sufficient amplitude and duration to fire or gate the silicon controlled rectifiers (SCR) in the positive lead of each battery. If the output of the optimum charge regulator (OCR) is in excess of the unregulated system loads (including boost regulator input current), the available current will be fed through the SCRs CR2 and CR4 into Battery 1 and Battery 2, respectively. The voltage on the anodes of the SCRs will be one SCR drop above the lowest battery voltage. If both batteries are at the same terminal voltage, both SCRs will conduct, allowing both batteries to charge. However, if the batteries are at unequal terminal voltage, the SCR for the highest voltage battery will be back-biased allowing all of the available charging current to go into the lowest battery in spite of both SCRs being gated.

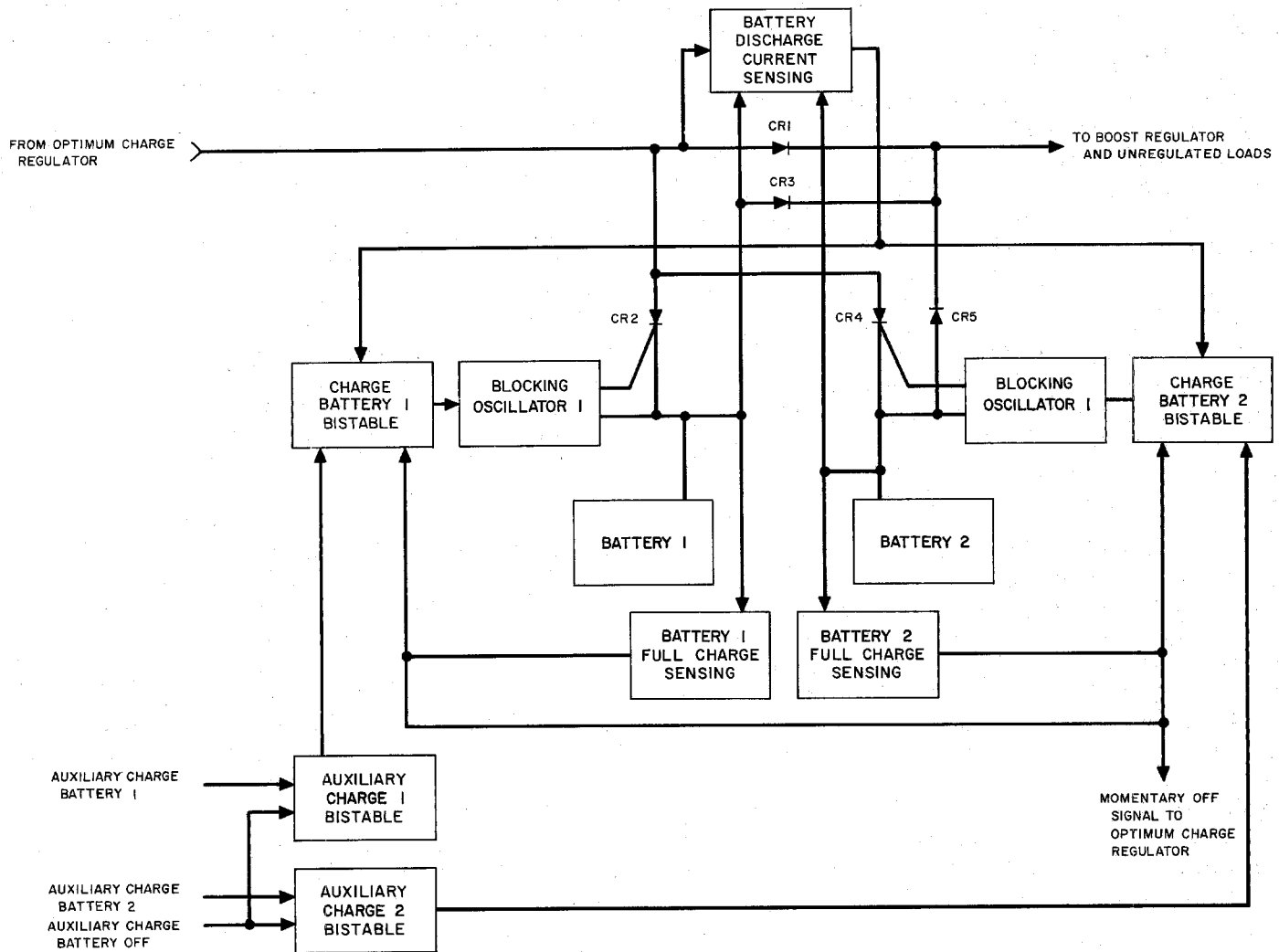


Figure 84. Battery charge logic functional diagram

When either battery becomes fully charged, a signal is sent from the respective "full charge sensing" circuit to the "charge bistable" and the OCR. The output of the OCR will momentarily (10-30 ms) reduce to zero while the blocking oscillator associated with the full battery will be inhibited by the output of the "charge bistable." When the OCR recovers, the SCR in series with the fully charged battery will be blocked allowing all the available current to be delivered to the other battery. When both batteries have been removed from the OCR, the output of the OCR is limited to approximately 28 v and supplies all load current within its capacity.

An auxiliary charge command will turn off the blocking oscillator associated with the battery not commanded "on." In addition, a pulse will be sent to momentarily

turn off the OCR. Thus the charging will be interrupted to the unselected battery and charging will be allowed on the selected battery. In this mode of operation, an output from the "full charge sensing" circuit will have no effect on the charging operation. This allows the command channel to override any automatic function.

c. Main overload trip circuit detailed operation. The principal elements of the main overload trip circuit (OTC) are the current sensing element and a power switch. In this circuit, current sensing is done by a tunnel diode (or parallel diodes as necessary) with a peak current point equal to the desired turn-off current. As long as the current through the tunnel diode remains less than the peak current point, the voltage across the tunnel diode is insufficient to cause conduction of the

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logic switch (Fig 85), and the power switch remains closed.

Power switching in the OTC is accomplished with a high-power silicon transistor which is held in saturation by the output of an electrical conversion unit (ECU). Since the ECU is self-starting, it will normally be on except when "inhibit" signals are delivered to the "off" terminal of the ECU. When an "off" signal is present, the ECU ceases to supply drive power to the switch transistor and it turns off.

If the current through the tunnel diode exceeds the peak current point, the tunnel diode voltage jumps to a relatively higher value. The associated logic switch detects this and causes an "off" signal to be delivered to the ECU. As the main transistor switch turns off, the current through the tunnel diode will fall below its valley current point causing the "off" signal to be removed. The ECU will then be allowed to turn back on causing the regulated voltage to again be available to system loads. The OTC must allow subsystem switches to automatically turn off or unlatch during the time the OTC is off. To accomplish this, a delay is incorporated in the self-starting feature of the OTC. This keeps the OTC off for greater than 20 ms before it can reapply the 29 v to the regulated bus.

If the regulated bus voltage falls below a preset level, the undervoltage sensing portion of the OTC causes a cycle of operation similar to that above except

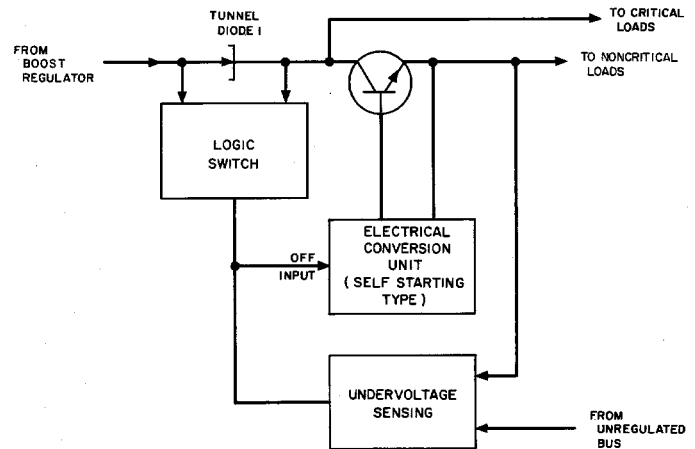


Figure 85. Main overload trip circuit functional diagram

that the noncritical loads cannot receive power until the bus voltages are above the preset minimum level.

d. Progress summary. Preliminary circuit releases have been made on the central power control circuitry. Final release of the entire unit awaits the result of temperature tests to be undertaken on the optimum charge regulator and the boost regulator. Battery charge logic and overload trip circuit development is completed except for decisions to be made on telemetering points and command functions in the charge circuit. Of the multitude of electrical conversion units and power switches, 90% of the circuits have been released.

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G. Thermal Control

1. Compartment Group

Thermally controlled compartments are used for spacecraft equipment which requires special thermal protection. The lunar day thermal performance of Compartments A and B has been refined to include the second-order lunar and solar inter-reflectance effects. Progress on the detailed thermal analyses of Compartments A and B, which include radiant interchange effects from all spacecraft surfaces in addition to lunar and solar inter-reflectances, is such that Compartment B is ready for computer study and the Compartment A detailed thermal model is in the final development stage.

Preliminary calculations indicate that at lunar noon the combined lunar and solar inter-reflectance effects degrade the compartment steady-state dissipating capability by 9 and 6 w in Compartments A and B respectively. The improved compartment dissipating capability obtainable with aluminized vycor mirrors having a solar absorptance, α of 0.12 and infrared emittance, ϵ of 0.85 as compared to the previous design value for switch radiators of α of 0.16 and ϵ of 0.8 is sufficient to offset the combined lunar and solar inter-reflectance effects. The current minimum compartment dissipating capability, using second surface mirrors for radiating surfaces and including lunar and solar inter-reflectance effects, is estimated to be 24 w in Compartment A and 12.5 w in Compartment B. These performance figures are for lunar

noon conditions with the solar panel-antenna assembly in its extended position above the compartment in question.

Figure 86 shows the steady-state dissipating capability of Compartments A and B from lunar noon to terminator. The transient capability of Compartment A during high-power transmitter mode operation is shown in Figure 87. Figure 87 indicates continuous high power transmitter operation is possible at lunar noon if the high-power mode, dissipating 37.5 w, operates on an effective 39% pulsed cycle basis in support of the television experiment. If the high-power transmitter mode dissipates 37.5 w on a 100% pulsed cycle basis, continuous operation is limited to 1.5 hr during a 4-hr period. This capability assumes an effective compartment heat capacity of 25 lb.

As previously reported, the lunar night Compartment A and B heat losses of 9.5 w (5.1 in A and 4.4 in B) were based on a combined wiring harness heat loss of 3.3 w. A detailed thermal study of the wiring harness exiting Compartments A and B using a nodal analysis and computer techniques has been set up and run successfully. Early results indicate that major problem areas exist with the heavy drill leads exiting Compartment B and the three coaxial antenna cables exiting Compartment A. Heat losses from these items alone were approximately 7 w, representing 90% of the total harness heat

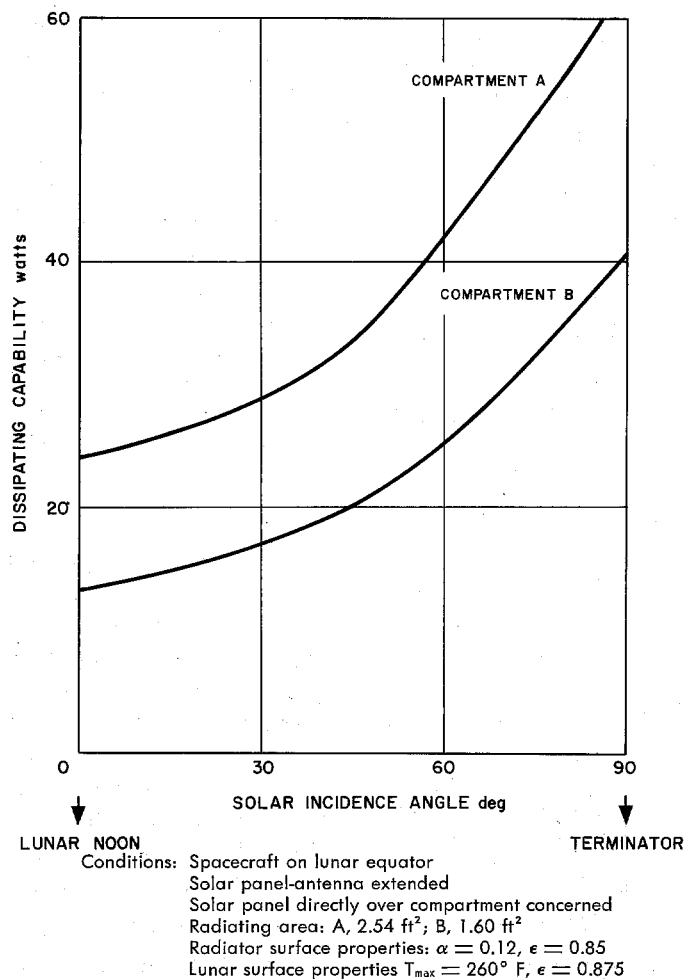


Figure 86. Compartment power dissipating capability

losses. Recent harness electrical and flexibility requirements have contributed to the large heat loss from the drill leads and coaxial cables.

The remaining leads, while large in number, have either been treated with a segment of nichrome wire or pass between Compartments A and B through a thermal tunnel; therefore heat losses from this group were already minimized to an acceptable level. A study of methods of reducing heat losses from the coaxial cables and the large power leads for the drill is under way. Assuming solution to the coax and drill lead problems, the current estimate of 9.0 to 9.5 w average power available throughout the lunar night for compartment thermal control remains fractionally less than current compartment steady state heat losses.

Since bulk compartment temperatures will be approximately 50° F at terminator, both compartments will have considerable transient capability as the spacecraft enters the lunar night. It is anticipated that the internal energy

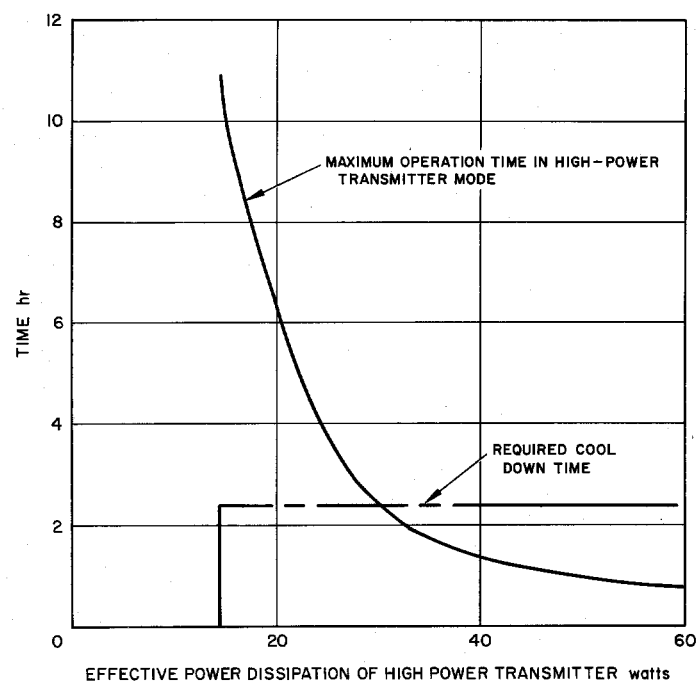


Figure 87. Compartment A transient power dissipation

expended during cooldown to 0° F will provide additional energy in sufficient quantity to ensure 0° F compartment temperatures through the lunar night.

Present mission sequencing information indicates it is undesirable to rely wholly on normal equipment dissipation during the night for compartment thermal control. Equipment reliability and possible RF interference with other space programs during DSIF down-time were major considerations. Electrical heaters will be installed on the lower portion of the thermal tray in both Compartments A and B. Each compartment will contain a diode-controlled heater capable of dissipating 10 w. The heaters will become operative when the thermal tray reaches 0° F due to average equipment dissipation levels less than 5 w. Each heater control unit contains only solid-state components.

A computer study of the thermal performance of Compartment C using two thermal control concepts has been completed. One concept employed a horizontal radiator integral with a vertical equipment mounting tray. The remaining surfaces were covered with $\frac{1}{2}$ in. of aluminized mylar superinsulation. The prime dissipator in this compartment is the subsurface sampler auxiliaries with 37 w for 15 min, cycled four times during the lunar day. With

this system the maximum equipment temperatures were less than 120°F during the period near lunar noon; however, equipment temperatures less than 0°F existed for the 90 hr preceding the day/night terminator with the compartment ultimately reaching a -90°F minimum temperature at terminator. Approximately 6 w auxiliary heating would be required for the 90 hr prior to terminator in order to maintain 0°F equipment temperatures. Because of the rather large power penalty, an alternative configuration was studied in which the equipment compartment was thermally coupled to a horizontal radiating surface with two thermal switches adjusted to close at 70°F. The prime function of the thermal switches in this application is to thermally decouple the equipment compartment from the radiator when the compartment cools below 70°F, thereby decreasing the compartment cooling rate. Compartment temperatures ranged from 120 to 34°F for environmental and equipment sequencing conditions consistent with the first study case. As a result of the study, this configuration will be integrated into the spacecraft design.

For T-4 and the SC series the thermal switch bodies will be offset from the geometric radiator center approximately $\frac{3}{4}$ in. on Compartment A and $\frac{1}{4}$ in. on Compartment B in order for a single radiator to be interchangeable with any switch body on a given compartment. In the MT-1 configuration, five radiator types were needed for Compartment A and two for Compartment B. This improvement will have negligible effect on the radiating efficiency of the switches.

2. Remote Subsystem Group

Preliminary calculations have indicated that some of the lower spaceframe tubes may experience transit temperatures of -65°F or lower. Since both the vernier and attitude control pressurization subsystems have a lower temperature limit of 0°F, the affected components must be thermally insulated from the spaceframe at all points of attachment.

The specific components involved are the vernier system helium tank, vernier system oxidizer tanks (three), vernier system fuel tanks (three), vernier engines, vernier system fuel and oxidizer lines, and altitude control pressurization tank (N_2). Insulation requirements of, in general, a maximum of 2 to 3 in² of a phenolic-type material $\frac{1}{4}$ in. thick have been specified and are currently being incorporated in the subsystem designs.

Investigations into methods of improving subsystem temperature control during transit with neither an increase in the auxiliary heating nor the introduction of a

lunar phase incompatibility have been completed. As a result, the spaceframe tubes below the compartment radiator level will be polished on the lower halves and painted with the low absorptance-high emittance white paint on the upper halves which receive solar illumination. This surface treatment has increased spaceframe temperatures from the previous estimates of -65°F maximum during transit (for illuminated portions). The low emittance of the polished portion ($\epsilon = 0.05$), combined with the white paint having a solar absorptance of 0.2 and an infrared emittance of 0.8, will yield spaceframe transit temperatures on the order of -25°F. Non-illuminated sections will be cooler (temperatures of -75 to -100°F are expected). No degrading radiant interaction effects with other spacecraft subsystems are expected during the lunar day.

Surface treatment is being applied to all vernier and attitude control system tankage subject to solar illumination during transit so that the non-illuminated portions will have an emittance of 0.05 while the illuminated portion will be striped in varying proportions with white and black inorganic paint to obtain the desired solar absorptance. An effective solar absorptance of approximately 0.30 and emittance of 0.8 on completely illuminated tanks will produce transit temperatures between 30 and 60°F. The partially shaded tanks will be striped with a pattern having a proportionally larger percentage of the black paint.

A two-dimensional thermal analysis of the main retro-engine was conducted to determine the propellant temperature history throughout transit. The effects examined were 1) solar heating, 2) case and nozzle emittance, and 3) structural support heat losses. The thermal model was a lune section which included an appropriate section of the nozzle, altitude marking radar, and structural supports. Propellant temperature at launch was assumed to be a uniform 80°F. Conduction and radiation effects from both the spaceframe at -65°F and local spacecraft equipment were included.

The configurations studied included the present gold case ($\epsilon = 0.04$), a case covered with $\frac{1}{4}$ in. of aluminized mylar superinsulation having an effective emittance of 0.02, the existing aluminum support, a redesigned support having a thermal conductivity of $k = 10$ Btu/hr ft °F (steel), and an insulated retro support/spacecraft joint. The results (Tab 15) are for no solar heating. The maximum and minimum propellant temperatures occur respectively at the star point at the engine head end and at the section adjacent to the case at the nozzle end. Solar heating on a gold case resulted in propellant temperatures of 350°F.

Table 15. Thermal study of main retro-engine

Structural support configuration	Propellant temperature after 66 hr, °F ^a			
	Gold case		¼ in. Aluminized mylar superinsulation	
	Maximum	Minimum	Maximum	Minimum
Present support (aluminum and steel)	51	13	58	21
With insulated joint	53	21	61	28
Aluminum support replaced with steel (k = 10 btu/hr ft °F)	51	19	59	28
With insulated joint	53	23	61	32

^a No solar heating; 80°F on-stand temperature.

Large annular shields to reduce the lunar environmental loading on the television cameras had previously been considered. This configuration introduced serious radiant interference with nearby spacecraft equipment by reducing the view to space required for thermal control of the affected items. An alternative method of providing thermal control for the television without causing thermal incompatibilities with surrounding equipment is currently being investigated. This method utilizes a smaller annular radiator (as opposed to a shield) located midway along the camera body. The radiator will result in both weight savings and reduction in radiant interference with other equipment.

Additional thermal analysis has been performed on the flight control group based on modifications in the design and on revised thermal radiation view factors. The results indicate that under steady-state transit conditions, the temperature of the mounting plate is approximately 25°F and that of the Canopus sensor approximately 0°F. In the previous design the mounting plate was at a higher temperature than the maximum operating temperature of the Canopus sensor (32°F), thus necessitating insulating mounts. It now appears possible to bolt the Canopus sensor directly to the mounting plate. More detailed analysis concerning this possibility is in progress.

Further analysis of the inertial reference unit has indicated that the radiator should be normal to the Sun's rays during transit. Such a geometry will minimize the heater power necessary during this phase of the flight. A second surface mirror is being considered for this purpose.

Since the current structural design of the doppler antenna dishes utilizes a complete fiberglass construction, additional thermal analyses have been made in a continuing effort to maintain the dish at a temperature above -40°F during transit. It has been shown that the minimum temperature of the all-fiberglass dish during transit is -117°F. To increase the heat conduction into the shaded portion of the dish, various thicknesses of aluminum foil over the top surface of the dish were investigated. The results are shown in Table 16.

Table 16. Thermal study of doppler antenna dish

Case	Configuration	Minimum temperature, °F
1	0.0007 in. aluminum honeycomb core 0.005 in. fiberglass cover sheet	-117
2	Same as above except 0.0007 in. foil cover on top surface	-70
3	Same as above except 0.0015 in. foil	-39
4	Same as above except 0.0033 in. foil	14

3. Scientific Instruments

The preliminary thermal analyses of most of the scientific instruments are nearing completion. Outside package surface temperatures for the lunar day operating conditions as well as the lunar night non-operating condition have been established. Latest lunar day estimates are given in Table 17. In addition, calculations have been performed to determine temperatures of the soil analysis group of experiments at near-terminator conditions. Minimum temperatures at the day-night terminator, heating requirements to bring instruments to operating values, and critical time in hours before terminator have been calculated. Critical time (θ_c) is here defined as the time in hours before terminator that an instrument passes through its minimum allowable operating temperature. Complete results are shown in Table 18.

A number of different techniques for maintaining the thermal environment of the radiation detector during the lunar night have been investigated:

- (1) The entire detector coated with a low emittance coating, mounted on the boom with a low conductance support, and supplied with electrical heating.
- (2) The entire detector wrapped in aluminized mylar insulation, mounted on the boom with a low conductance support and supplied with electrical heating.

- (3) The electrical portion of the detector isolated from the ionization chamber, with low emittance coating and heated electrically.
- (4) The electrical portion of the detector isolated and wrapped in aluminized mylar insulation and heated electrically.
- (5) The electrical section enlarged to allow the electrical circuitry to be wrapped in aluminized mylar, supported within the case on low conductance supports and heated electrically.

Table 17. Scientific instrument daytime outer surface temperature

Experiment	Package	Dissipating surface maximum operating temperature, °F
X-ray diffractometer	Head	325
	High-voltage power supply	190
X-ray spectrometer	Head	160
	High-voltage power supply	170
Gas chromatograph	Package	170
Radiation detector	Electronics	260
Surface geophysical	Interferometer	129
	Thermal diffusivity	170
	Acoustic source	260
	Penetrometer	260
	Magnetic susceptibility	260
	Acoustic sensor	260
	Density	260
Subsurface geophysical	Density	200 — 260
	Thermal diffusivity	200 — 260
	Temperature	200 — 260
	Magnetic susceptibility	200 — 260
	Caliper	200 — 260
	Acoustic sensor	200 — 260
	Hole wall temperature	260
Soil mechanics	Body	160

Table 18. Thermal study of soil analysis group

Instrument	Specified minimum operating temperature, °F	Temperature at terminator, °F	Critical time, θ_c , hr before terminator	Auxiliary heat required for operation, w
Spectrometer head	-40	-191	51	13.7
Spectrometer high-voltage power supply	-40	-183	29.5	4.03
Diffractometer head	-40	-144	29.5	31.5
Diffractometer high-voltage power supply	-40	-183	29.5	6.1
Gas chromatograph	32	-146	59.2	1.51

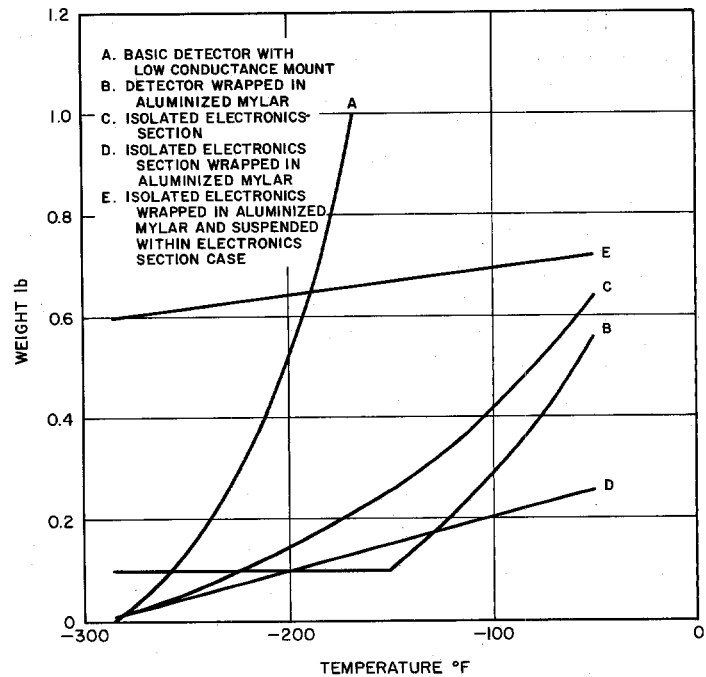


Figure 88. Thermal control of radiation detector

Figure 88, typical of the results of the analysis, shows the thermal control weight necessary to achieve a particular minimum night temperature for the various techniques outlined above. The weight estimate includes battery, insulation and additional weight due to dimensional changes.

All the surface and subsurface geographical experiments have been reviewed. Only two of the surface experiments have been analyzed in detail. The others, due to their extreme ruggedness and temperature insensitivity, will not require detailed thermal analysis. Results of the total-temperature interferometer thermal analysis indicate the instrument will operate well within its allowable temperature during the day. For nighttime operation, a heater of about 2 w capacity will be required to bring the instrument to operating temperature. This requirement has been recommended to the vendor. Preliminary analysis of the thermal diffusivity experiment indicates that the interferometer used in this experiment will also require a 2 w heater for nighttime operation.

In the subsurface geophysical experiment, all the equipment on the logging sonde has been reviewed and does not appear critical with the possible exception of two items: the subsurface temperature sensor and the downhole interferometer. Analysis of the subsurface temperature sensor indicates the sensor will operate at a reasonable temperature, but more analysis is required

to determine the effect of possible sensor selfheating. Although not as yet studied in detail, it appears that the interferometer on the logging sonde may operate at a temperature in excess of its maximum allowable temperature of 170°F, especially at shallow lunar depths. Further analysis is planned.

A simplified mathematical model has been analyzed to investigate the feasibility of using an insulation blanket around the seismograph to minimize transverse surface temperature gradients. The model analyzed was a two-dimensional semi-infinite slab insulated at the upper surface, maintained at a constant temperature at the lower surface, and subjected to a periodic depth-dependent forcing function at another face (Fig. 89). The steady periodic solution indicates the insulation should be deployed a distance $X = 0.3D$ from the base of the seismograph in all directions, where D is the penetration depth. The results of these analyses, although valid and complete in themselves, will be used as a point of departure for a more detailed numerical study with more realistic boundary conditions.

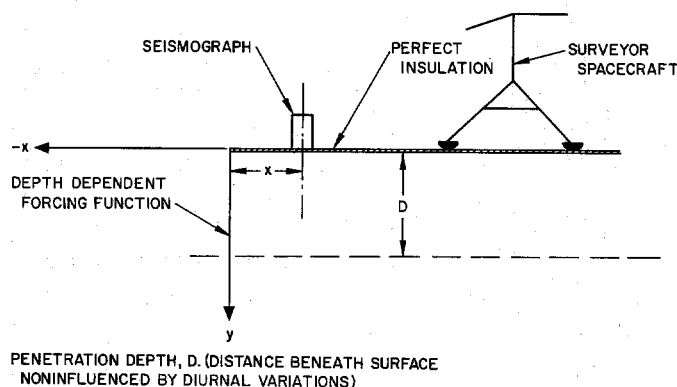


Figure 89. Model for seismographic-lunar surface insulation analysis

The two lunar surface models specified for use in the thermal design of the seismograph—the homogeneous dust and homogeneous rock models—have been re-examined. The resulting equatorial surface temperature-time histories are shown in Figure 90. Former results reported in the previous Bimonthly Progress Summary were for two non-equatorial locations and should hereafter be disregarded in favor of the profiles shown.

4. Thermal Switches

Recent testing with copper-plated contacts indicates an over-all switch conductance of 0.45 Btu/hr°F as opposed to the value of 0.35 Btu/hr°F previously reported. The

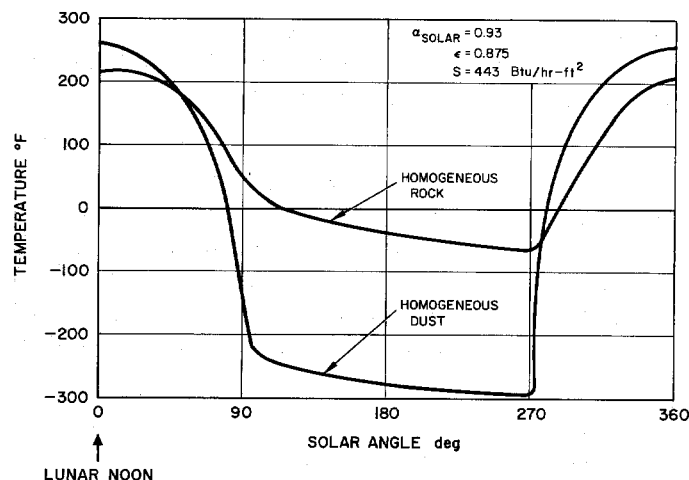


Figure 90. Lunar equatorial surface temperature

increased conductance was realized by more precise initial contact parallelism.

Although copper offers a minimum of difficulty in both plating and lapping, it appears that the promise of increased contact performance (approaching the desired value of 0.7 Btu/hr°F) with gold outweighs these advantages. Gold, in addition to its increased thermal performance, offers a more chemically stable surface: the gold will not suffer from surface oxidation as would copper in both the heat sterilization and prelaunch environments. A group of prototype thermal switches are presently being fabricated for use on the compartments of the MT-1 test sectors. These switches are to have gold contact surfaces.

Thermal switch cases 10 mils thick have been successfully machined from solid Rexolite bar stock. Tests are under way to verify the thermal conductivity and determine the creep rate under load at the sterilization temperature. If the test results are favorable, this material will meet all requirement for the switch case. The investigation of Ablatalite is still in progress. However, machining difficulties have arisen and until a less brittle compounding of the material can be obtained, the qualification of Rexolite will be given major emphasis.

5. Compartment Test

Three thermal tests of the compartment development model (described in the previous Bimonthly Progress Summary) in a simulated lunar night environment were completed during January. At least three additional test phases are now planned for the near future.

The ambient pressure within the chamber throughout these tests was between 3 and 5×10^{-6} mm Hg, and the

effective wall temperature was between -200 and -210°F . Operation of both the vacuum and cold wall systems was on a continuous 24-hr basis.

The insulation configuration in the first test consisted of 100 sheets of $\frac{1}{4}$ mil aluminized mylar with all joints completely interleaved. The lacing thread used to contain the mylar during the assembly process was not removed after assembly of the insulation blanket into the canister. No special wrapping techniques were attempted around the structural ties between the inside and outside canister.

The initial cooldown data with no internal power dissipation are shown in Figure 91. Once the thermal tray temperature reached 0°F , power was applied through heater blankets bonded to the tray. A 20°F thermal tray equilibrium temperature was reached with a total internal power dissipation of 3.4 w. A heat balance based on the experimental data yielded a net loss of 0.15 w/ft^2 through the insulating blanket, or 0.14 w/ft^2 when adjusted for an internal temperature of 0°F and the expected outer wall temperature of -300°F . The thermal model being tested, however, has significantly less total heat capacity than the ultimate compartment; hence the cool-down rate is not directly applicable to the final design.

In the second test, an attempt was made to evaluate the effect of overlapping rather than interleaving the mylar sheets around the top joint. The overlapped configuration would afford easier penetration of the ethylene oxide for sterilization, and the final assembly time could be cut from 2 to 3 hr down to a matter of minutes. Although a degradation in performance was observed in this test, the quantitative results are questionable due to operating difficulties with the vacuum chamber-cold wall

facility (e.g., loss of vacuum and cold wall). This phase is to be rerun at a later date.

The third phase again utilized the original interleaved configuration. Following the first test, it was suspected that the tension in the lacing ties had created localized areas of high contact conductance between sheets. The lacing thread was therefore removed, leaving the insulation blanket to be supported by the canister wall and the internal wire mesh. The original plan for this phase had been to obtain several steady-state data points. The compartment was allowed to cool from room temperature to 30°F and 4.2 w power was applied. Based on an extrapolation of the Phase 1 data, this power level should have resulted in an equilibrium of 45 to 50°F ; however, the compartment reached equilibrium at 70°F more than 3 days later (Fig 92). Only one additional steady-state point was obtained before conclusion of the test period. Although this second point was reached near termination of the test and may not represent a true steady-state situation, it is significant because of the manner in which it was obtained. The power was removed at 104 hr, as indicated in Figure 92. At 110 hr the tray temperature of 44°F was noted and, using an extrapolated power-temperature characteristic (from the family of analytically generated power curves shown in Figure 93) the power setting for equilibrium at this new temperature was determined. The power applied to the compartment was 3.7 w, or approximately 0.10 w in excess of that required by the extrapolation. The fact that the tray temperatures tended to increase as indicated in Figure 92 implies that the insulation effective emittance is equal to or less than 0.0058.

The significant increase in performance of the insulation over that obtained in Phase 1 is attributed to decrease in sheet-to-sheet contact conductance through

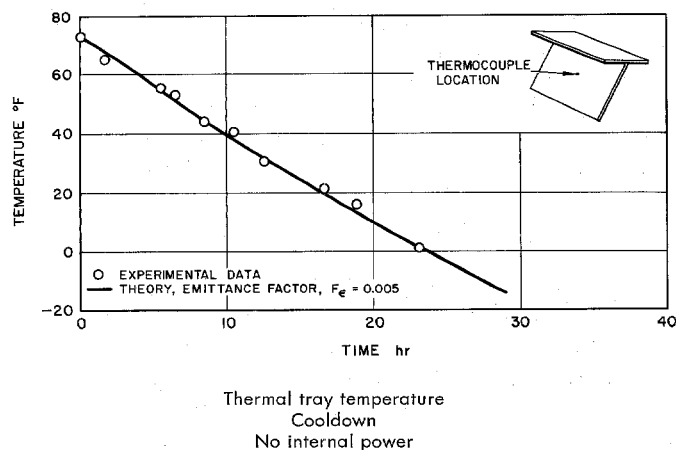


Figure 91. Compartment thermal test data

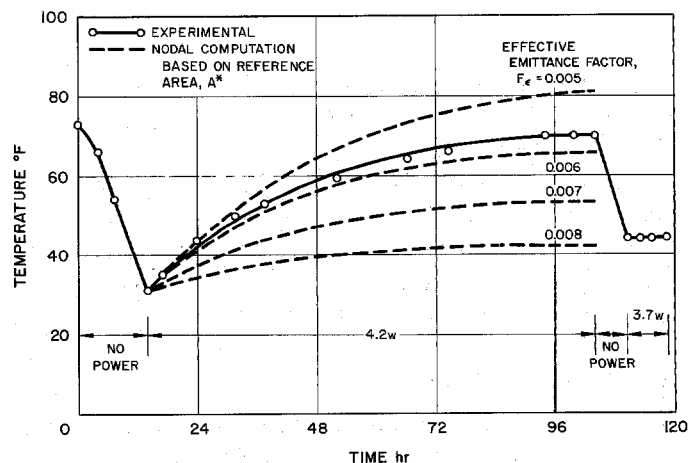


Figure 92. Tray temperature in compartment test

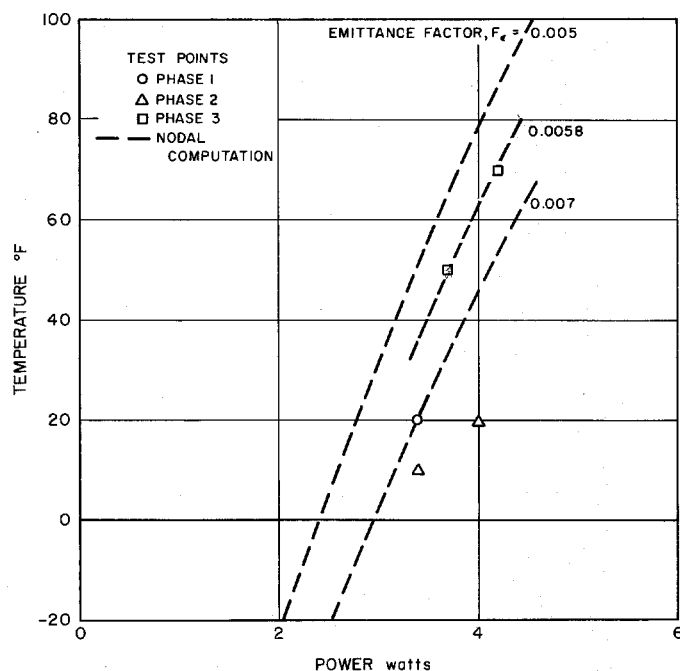


Figure 93. Steady-state power-temperature characteristics

removal of the lacing ties. The only other reasonable explanation would be a variation in the contribution of gas conduction in Phases 1 and 3. The analytical curves for the transient shown in Figure 92 compare quite well with the experimental curve shape and provide strong evidence that no variation in performance occurred due to a slowly decreasing gas pressure in the insulation. It is concluded that if outgassing were affecting the performance of the insulation, its effect was relatively constant during the test period. Since the ambient pressure within the vacuum chamber was nearly identical for both Phases 1 and 3, it is also concluded that gas conduction effects, if present, did not vary significantly between phases.

The data from Phase 3 results in a heat loss value of 0.115 w/ft² of outside surface area when adjusted to the lunar ΔT of 0 to -300°F, which compares very closely with the current design value. The fourth phase is to be a continuation of Phase 3 for the purpose of establishing the power-temperature characteristics of the box over a wide range of temperatures.

At least two additional phases are now being planned. In the fifth phase, the number of shields is to be reduced to 75. The final phase will constitute a rerun of Phase 2 in order to establish the degradation in performance due to overlapped joints. This phase will consist of either the 75- or 100-sheet configuration, depending on the results of Phases 4 and 5.

Up to the present time, all design calculations involving the use of multiple-layer insulation have been based on a somewhat arbitrary selection of an emittance factor. The outside surface area of the compartments has been used in conjunction with the emittance factor in the design calculations of insulation heat loss. With the prospect of using an experimentally determined value of the emittance factor, it becomes necessary to take into account the differences in compartment configurations; since for a compartment of finite size, the insulation area is not constant with the thickness. In the future the following relationship for the effective insulation area will be used in conjunction with an experimental value of emittance factor to determine the heat loss from any compartment face:

$$A^* = A_i \left\{ \frac{1}{\Delta l \left(\frac{C_i}{A_i} \right)} \ln \left[1 + \Delta l \left(\frac{C_i}{A_i} \right) (N + 1) \right] \right\}$$

where

A_i = inside surface area of side

Δl = one-dimensional increase in length per sheet

C_i = total circumference of inside sheet

N = total number of sheets

6. Coatings

Radiation coating selections for the MT-1 thermal test sectors have been established. It is anticipated that, in general, the MT-1 coatings will be the recommended coatings for both T-4 and the SC series.

Seven coatings or coating combinations have been selected:

- (1) High power dissipating surfaces — second surface vycor mirrors; $\alpha = 0.12$; $\epsilon = 0.85$.
- (2) Low power dissipating and general upward looking surfaces — white inorganic paint; $\alpha = 0.244$; $\epsilon = 0.956$.
- (3) Low emittance surfaces — polished aluminum; $\alpha = 0.3 - 0.4$; $\epsilon = 0.05$.
- (4) Low emittance surface — aluminum foil; $\alpha = 0.19$; $\epsilon = 0.06$.
- (5) High emittance — no solar requirement — black anodize; $\epsilon \geq 0.8$.
- (6) High solar absorptance — high infrared emittance — inorganic black paint; $\alpha = 0.966$; $\epsilon = 0.929$.

- (7) Low emittance — low solar absorptance — vapor deposited aluminum; $\alpha = 0.1$; $\epsilon = 0.05$.

Second-surface vycor (No. 7910) mirrors have continued to demonstrate power dissipating properties (low solar absorptance, high infrared emittance) superior to any other materials investigated. The remaining problem is bonding the plated mirror to the aluminum thermal switch radiators with a high thermal conductance bond. Since the mirror and switch radiator plate assembly will experience temperatures ranging from 100°F during the lunar day to -320°F during the lunar night, the differences in the two coefficients of thermal expansion can cause high radial stresses in the Vycor. To partially relieve this problem, the current design concept utilizes a four-plate mosaic on each switch radiator. The individual mosaics will be on the order of 2 to 3 in. on a side. The most promising bond appears to be a solder bond using an indium-aluminum solder. The mirror would be prepared by depositing gold onto the evaporated aluminum layer. The gold plate will be on the order of 5000 to 6000 Å thick. The aluminum switch plate will be similarly gold-plated and sweat-soldered in place. Initial tests of individual plates on the order of 2 to 3 in. on a side have shown no thermal cracking when submerged in liquid nitrogen.

Preliminary data from UCLA indicate that the white inorganic paint has a solar absorptance of 0.244 and an infrared emittance of 0.956. While the solar absorptance is somewhat higher than anticipated, the infrared emittance exceeded the design goal of 0.8, and therefore the α/ϵ ratio of 0.25 used for design purposes is still achieved. It is felt that the slightly higher solar absorptance is due to an insufficient layer of paint; hence further testing using an increased number of thickness coatings is planned.

Preliminary UCLA data on the inorganic black paint indicate a solar absorptance of 0.966 and infrared emittance of 0.929. This paint, in conjunction with the white (stripes or polkadots), will be used to obtain effective solar absorptance values between 0.25 and 0.9.

Because of its high degree of reproducibility, polished aluminum has been selected for low emittance applications. While gold has a somewhat lower emittance, initial work at JPL indicated large variations between samples and, in addition, some changes subsequent to surface cleaning or buffing. The required aluminum coatings will be obtained by one of three techniques depending on the application. If the base material is of aluminum and can be polished and buffed, no further coating is required. If the base material cannot be polished, either (1) a

layer of aluminum foil will be applied (as over vernier lines and heaters) or (2) the surface will require a vapor-deposited aluminum over an epoxy substrate. Indications from the literature are that the latter method results in a somewhat lower solar absorptance (0.1 vs 0.2 to 0.4).

Black anodize has been specified for all components within the major compartments.

7. Full-Scale Thermal Model Tests (MT-1)

The initial MT-1 Sector 3 test is scheduled to begin on April 6. All thermocouple locations (approximately 150) have been selected and the power control unit is under construction. This first test of Sector 3 will be under lunar night conditions. The items of particular interest during this phase will be: the gas chromatograph, the thermal diffusivity instrument and the surface temperature sensing instrument, since these units represent the only lunar night power dissipators.

Both the sector power control unit and the temperature instrumentation to be used in the night tests of each sector will be sufficient for the needs of the later transit and lunar day tests. Hence, no major modifications should be required in preparing the sectors for testing under solar conditions. The transit and lunar day tests are to begin in early June.

8. Engineering Temperature Sensors

To record engineering temperature data, 41 temperature sensors have been located throughout the basic spacecraft. Electronic considerations have led to the selection of resistive type sensors having a direct 0 to 5 v output. The temperature data obtained will be used for thermal control system performance verification, system malfunction diagnoses, and subsequent spacecraft thermal performance predictions. The common accuracy requirement is set at $\pm 2^\circ\text{C}$. The locations of the various sensors are indicated in Table 19.

9. Lightweight Surveyor — A

Preliminary feasibility studies of a lightweight spacecraft configuration (2100 lb) have indicated that transit and lunar day thermal control of Compartments A and B will remain unchanged and will provide temperatures between 0 and 125°F. This configuration, while offloading, among other items, one 45 lb battery from Compartment A and the complete Compartment C, has resulted in the creation of a new compartment — D. Compartment D, housing the auxiliary 5 lb battery and the tape recorder, would have two thermal switches set at a nominal

Table 19. Temperature sensor location

Flight control (1) Electronics unit (2) Electronics unit (3) Inertial reference unit (4) Canopus sensor (5) Nitrogen tank (6) Altitude marking radar dish (7) Altitude marking radar electronics (8) Altimeter and doppler radar klystron unit (9) Altimeter and doppler radar signal data converter (10) Altimeter or doppler radar dish Electronics (11) Transmitter A (Compartment A) (12) Transmitter B (Compartment A) (13) Power control system (Compartment B) Electrical power (14) Battery No. 1 (Compartment A) (15) Battery No. 2 (Compartment A) (16) Solar panel Mechanisms (17) Planar array antenna Thermal control (18) Compartment A — thermal tray — top (19) Compartment A — thermal tray — bottom (20) Compartment A — thermal shell — inside	(21) Compartment A — thermal shell — outside (22) Compartment A — thermal switch — inside (23) Compartment A — thermal switch — outside (24) Compartment B — thermal tray — top (25) Compartment B — thermal tray — bottom (26) Compartment B — thermal shell — outside (27) Compartment B — thermal switch — outside (28) Compartment C — thermal tray (29) Compartment C — thermal switch — outside Propulsion (30) Main retro case (31) Vernier lines (32) Vernier tank (33) Vernier tank (34) Vernier motor Vehicle (35) Upper spaceframe (36) Lower spaceframe (37) Wiring harness — thermal tunnel (38) Shock Absorber 1 (39) Landing Gear 1 Television (40) Vidicon — Camera 4 (41) Electronics — Camera 4
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70°F contact closure temperature. The compartment would require 2 to 3 w heating during transit and provide 0 to 125°F temperatures during both transit and lunar day. The compartment would not be thermally controlled during the lunar night and would therefore experience temperatures in the vicinity of -300°F. Depending on the survival capability of the enclosed equipment, the compartment may have second day capability.

Assuming that 2450 w-hr of battery energy (45 lb) are available for lunar night compartment thermal control, a lunar night mission sequence making maximum use of the compartment transient response has been established. This sequence, using the minimum amount of power required to maintain an RF link during the transient period prior to reaching a 0°F compartment equilibrium temperature, will yield a maximum output of engineering data on the compartment thermal control system. Figure 94 shows the cooling rate of the compartment equipment from day/night terminator. The compartment will contain auxiliary heaters which can, if desired, automatically turn on at 0°F and will supply the 5 w power required to control each compartment to 0°F. Based on the 2450-w-hr battery capability, a minimum of 200 hr of 0°F thermal control is assured during which engineering data for evaluation of the thermal control performance can be gathered. Subsequent to this period, Compartment A and B equipment will cool to -50 and

-115°F, respectively. Initial test results indicate that the compartment equipment will survive these temperatures, and at sunrise electrical power from the solar panel will be available to reheat the equipment. The logic of the power control and heater systems will bring the heaters into use as soon as the solar panel is energized. Compartment A will reach 0°F approximately 32 hr after sunrise; similarly, Compartment B equipment will reach 0°F 28 hr after sunrise. Nominal operation of the basic bus electronics is therefore possible 32 hr after sunrise.

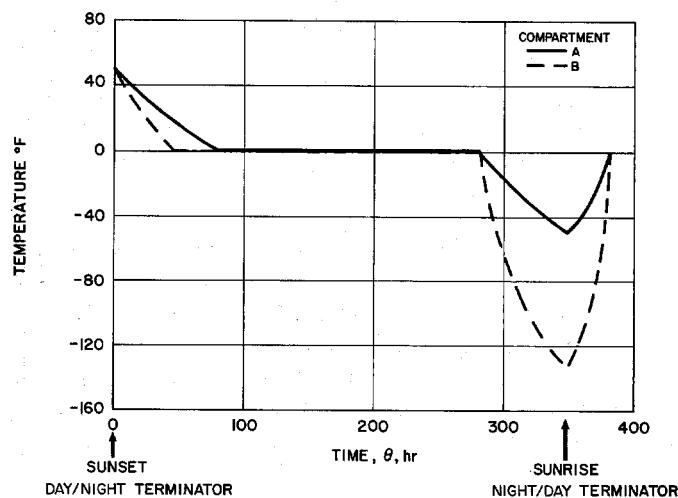


Figure 94. Compartment temperature during lunar night

H. Engineering Mechanics

1. Touchdown Dynamics

In the previous Bimonthly Progress Summary, Figure 116 showed the package acceleration contours for the case of a 20 ft/sec landing. For the same parameters at a touchdown velocity of 15 ft/sec, the acceleration values are essentially constant over the complete region covered by the 10σ dispersion ellipse and do not exceed 22 g.

A more detailed investigation of the effect of variations of touchdown parameters such as touchdown weight, landing attitude, radius of gyration, and vertical center of gravity^a has been carried out. Figure 95 indicates the effect of weight variation on solar panel-antenna lateral acceleration. It is noted that a $\pm 5\%$ variation in touchdown weight does not appreciably affect the inertia loads on the spacecraft. Further work is being carried out to ascertain the effect of other parameters.

Friction coefficients above 1.0 gave the stability boundaries as shown in Figure 118 of the previous Bimonthly Progress Summary. Additional investigation of the effect of friction on stability has resulted in Figure 96. This characteristic of the stability boundary moving to higher vertical velocities without much change in lateral velocity as friction is reduced has also been found on a 25 deg site slope.

^aThis vertical center of gravity is measured above the landing gear hingeline (Station 47.48) and is denoted as Δcg .

If, regardless of vertical touchdown velocity, the smallest lateral velocity which results in toppling for a given set of conditions is denoted as the maximum allowable lateral velocity for those conditions, this velocity is a conservative indication of stability. Figure 97 is a plot

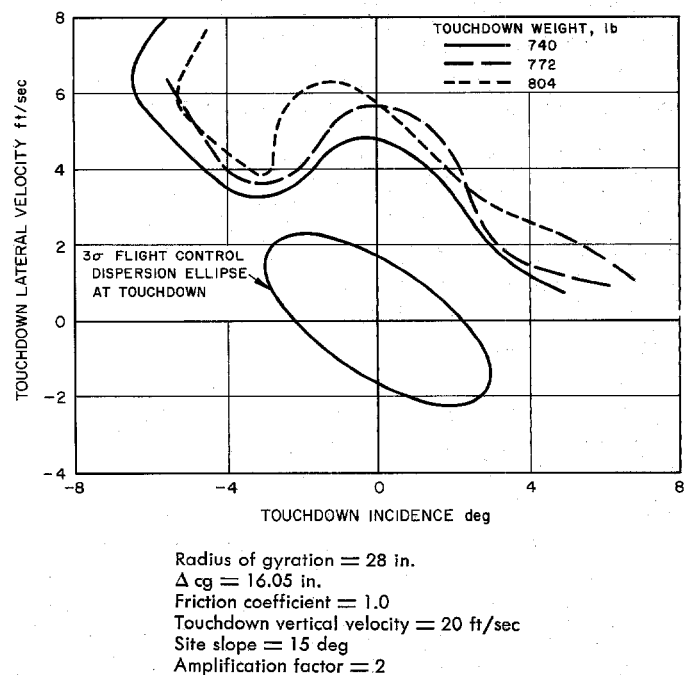


Figure 95. Antenna lateral acceleration vs touchdown weight

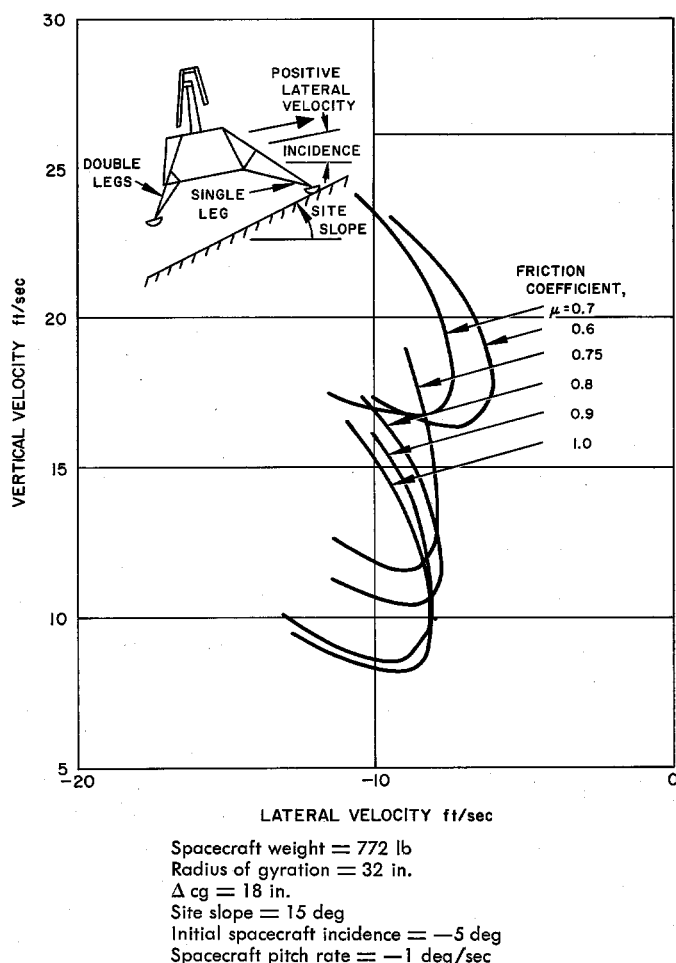


Figure 96. Effect of friction on stability

of this maximum allowable lateral velocity as a function of pitch radius of gyration and Δcg . This curve indicates that the lateral velocity capability on a 15 deg site slope is decreased by about 1 ft/sec for each 2 in. increase in the center of gravity height. If the pitch radius of gyration is held constant at 32 in., and the maximum

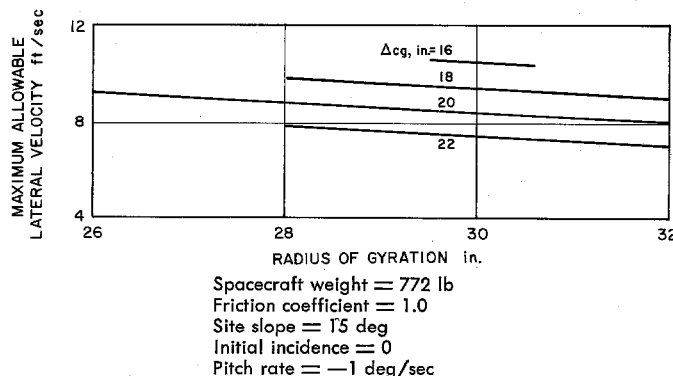


Figure 97. Effect of radius of gyration on lateral velocity

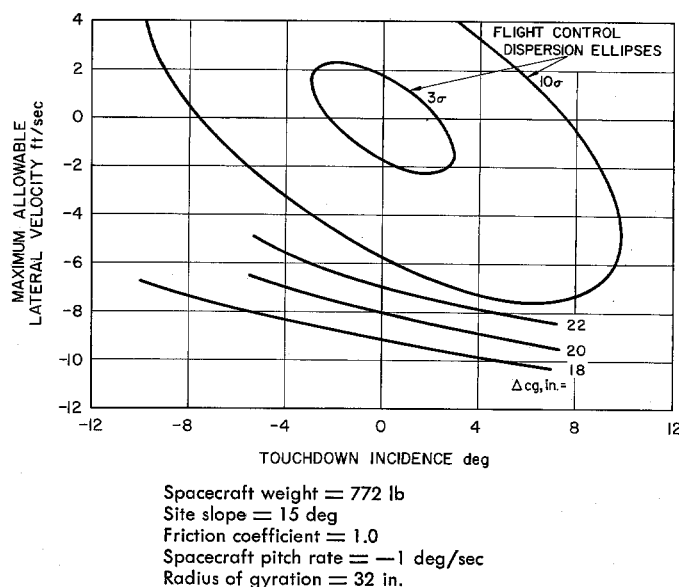


Figure 98. Effect of initial incidence on lateral velocity

allowable lateral velocity is found for various spacecraft attitudes at touchdown, the result is as shown in Figure 98. The criteria for acceptable center of gravity height has been established on the basis of these data as well as retaining as high a maximum slope capability as possible. As long as the boundary defining maximum allowable lateral velocity is outside the 10σ flight control dispersion ellipse, the stability on a 15 deg site slope is considered acceptable.

The site slope for which the maximum allowable lateral velocity just penetrates the 3σ dispersion ellipse is denoted as the maximum site slope capability.

Present indications are that for a Δcg of 18 in., the maximum site slope capability is about 28 deg. The effect of slopes in excess of 15 deg is shown in Figure 99. Figure 99a shows the progression of stability boundaries with site slope for a Δcg of 18 in. while Figure 99b is for a Δcg of 20 in. The development of a second knee in the stability boundary at higher vertical velocities has become pronounced only as higher center of gravity conditions have been investigated. This second knee has a very unstable characteristic with increases in site slope for high vehicle center of gravity locations, as shown by Figure 99b.

Investigations of touchdown dynamics for a soft "dust-like" lunar surface have begun using the soil mechanics definitions^a called out in JPL Specification 30240 A. The

^aA soft inelastic material with high internal friction having a compressive strength equal to zero at the surface and increasing linearly with the depth of vertical penetration at a rate of 10 lb/in²/ft.

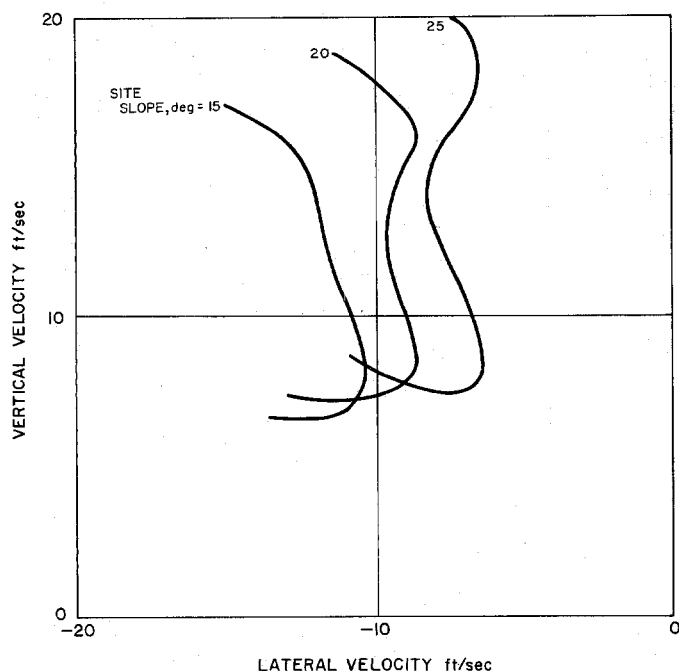
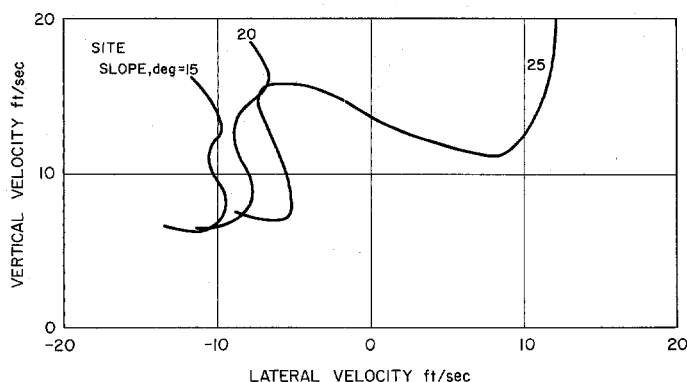
a) $\Delta cg = 18$ in.b) $\Delta cg = 20$ in.

Figure 99. Effect of site slope

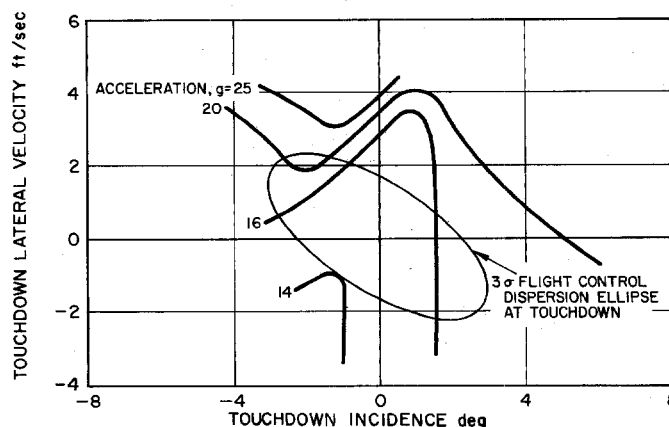
landing gear foot pads will penetrate the lunar surface, and the resulting forces are a function of depth of penetration. This led to extensive modification of the existing computer program, for which the rewriting has been completed. This program balances the shock absorber force against the resistance force of the surface in order to determine the rate of leg deflection. This simulation is necessary since the penetration for a normal impact can vary between about 8 and 18 in., depending on the amount of leg deflection considered.

The vertical force on the foot pad is determined by applying the specified pressure at the depth of pene-

tration to the foot pad horizontal cross-sectional area. This same pressure is used to determine part of the transverse force on the foot pad by applying it to the projected vertical cross-sectional area. An additional transverse force is applied in the form of soil friction as used in land locomotion problems.

There is also a pressure force on the rigid arm portion of the leg which is proportional to the variation of penetration along the length of the arm. This force is proportionately applied to the foot pad pivot point and the leg pivot point at the spaceframe. With the forces applied to the spaceframe from the rigid arm, crushable block, and shock absorber, the motion of the assumed rigid spaceframe is calculated as in the previous IBM program. This analysis compares with the original program where the surface resisted with whatever force was necessary to counteract that supplied by the shock absorber. It is expected that the problem will have been programmed and checked out by the latter part of February.

Some exploratory investigations for the effect on loads and stability of a 25% lighter landed weight vehicle have been conducted. Preliminary results indicated as much as 50% higher lateral loads on the antenna-solar panel group. Therefore, it was considered necessary to reduce the crushable body block force and shock absorber damping coefficient in proportion to the weight to retain the same level of inertia loads. The result of reducing the input forces is shown in Figure 100 which is comparable to Figure 114 of the previous Bimonthly Prog-



Spacecraft weight = 579 lb
 Radius of gyration = 28 in.
 $\Delta cg = 18$ in.
 Friction coefficient = 1.0
 Touchdown vertical velocity = 20 ft/sec
 Site slope = 15 deg
 Amplification factor = 2

Figure 100. Antenna lateral acceleration

ress Summary and demonstrates a slight reduction in inertia loads. The reduction in the damping coefficient also results in lower shock absorber loads being applied to the spaceframe.

The reduction in damping coefficient without a change in spring constant gives a relatively stronger spring, which results in slightly better stability characteristics. Figure 101 shows stability boundaries for a given set of initial conditions in the case of the lighter design as compared to the present design. Although only this one stability boundary has been investigated, the comparison should be representative of the results for variations in friction, center of gravity, site slope, etc. In terms of the effect of center of gravity height on stability for the present design, the increase in stability of 1 ft/sec should permit about a 2 in. increase in vertical center of gravity height for a lightweight vehicle relative to the present design to give the same stability.

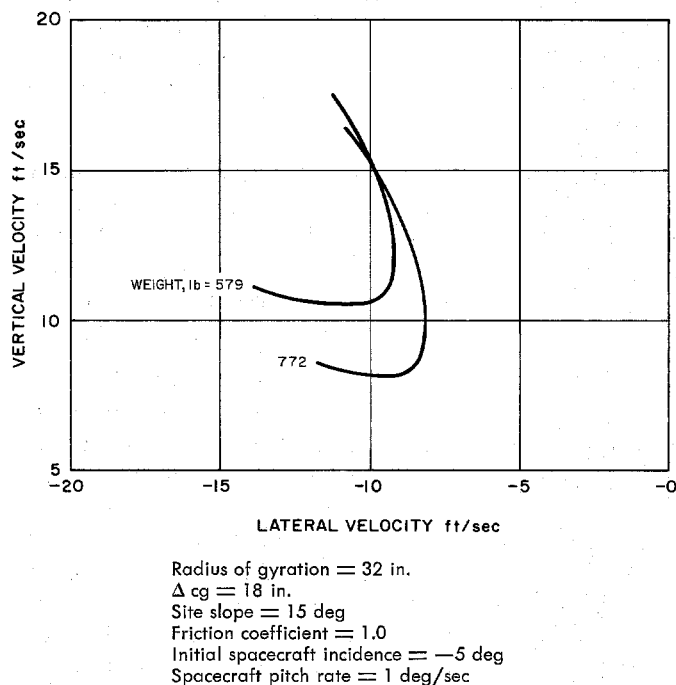


Figure 101. Effect of weight on stability boundary

2. Full-Scale Drop Tests (T-1)

The preliminary test plan discussed in the previous Bimonthly Progress Summary has been finalized. Five additional drops have been added (bringing the total to 30) in order to give better coverage to the stability tests. Instrumentation will consist of high-speed motion



Figure 102. Drop test facility

pictures, 13 accelerometers, 45 strain gage bridges, deflection potentiometers on each shock absorber, and a load cell to monitor the "anti-gravity" force.

The upper half of the 15 deg drop platform has been erected. The flooring consists of 10 by 12 in. wooden beams supported by 12 by 12 in. wooden columns. Figure 102 shows the completed tower and upper landing platform. The lower half of the platform will be completed after the first T-1 drop test, onto a level surface.

The lateral velocity pendulum (also shown in Figure 102) has been completed and installed. A mechanical trip mechanism for releasing the spacecraft from the pendulum has been built, tested, and installed on the pendulum. A trip mechanism for releasing the pendulum has also been assembled and tested. Tests of the mechanical system have been performed using a test speci-

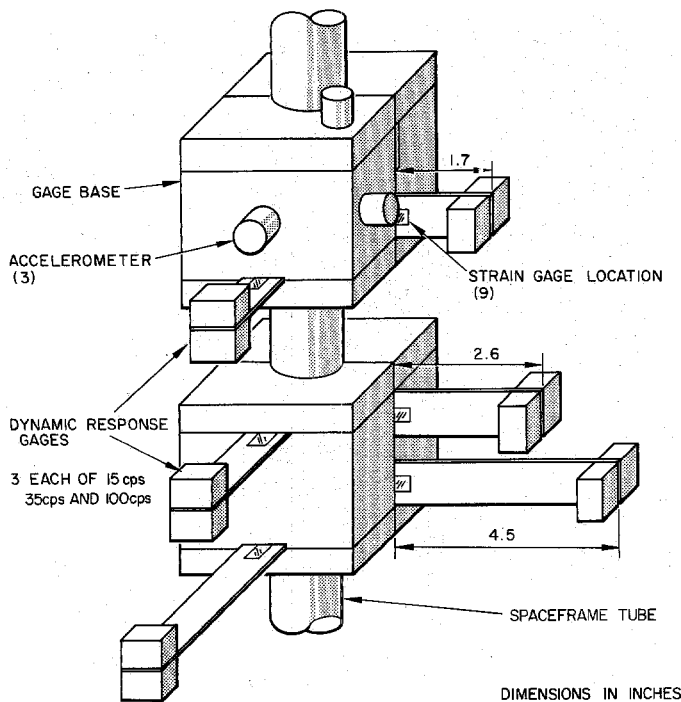


Figure 103. T-1 structural response indicators

men simulating the spacecraft touchdown weight. All parts of the system functioned satisfactorily during these tests.

Estimates of dynamic structural response to landing shock will be obtained from dynamic response indicators mounted on the spacecraft (Fig 103). These indicators are designed to respond at different frequencies to indicate the frequency content of the shock loads imposed on the structure at landing. The assembly will be mounted on the vertical post of the spaceframe structure which is above the crushable blocks.

The dynamic response indicator consists of a set of three reed gages and an accelerometer for each of the x, y, and z axes of the spacecraft. The set of three reed gages are tuned at 15, 35, and 100 cps to provide three points for a shock load spectrum. Recorded data is from a strain gage at the base of each reed. At present the data will be examined for the peak response only. The accelerometer accompanying the set of three reed gages is a backup instrument to provide a detailed input time history, should it be required. The accelerometer data will be recorded on magnetic tape; the strain gage data, on oscillograph paper.

The over-all capability of the unit — and in particular the directional capability — will be determined by a calibration experiment. In this experiment the unit will

be mounted on a tube that simulates the post of the spaceframe. The tube will be rigidly attached to a fixture and subjected to controlled shocks in each of the three sensitive directions. The response of the accelerometers will provide data to determine the shock spectra to compare with the approximate shock spectra obtained from the three reed gages of the set.

In addition to the dynamic response indicators, 18 strain gages will be attached to the spaceframe structure to obtain axial loads in the tubes. An additional 18 strain gages will be installed on the landing gear structure.

3. Shock Absorber Development

The development model of the shock absorber has undergone testing to define the fluid volume necessary for the liquid spring and the metering rod groove profile associated with the damping. The test setup, indicating the test shock absorber, load cell, and driving cylinder, is shown in Figure 104. Dynamic testing of six metering rods with different groove profiles has led to the final configuration selection for this part. The final spring and damping curves have been established within the specification limits.

Difficulty has been experienced in obtaining a tapered titanium tube for the column portion of the shock absorber. It has been found desirable in terms of schedule and reliability to use a straight tube. Therefore the prototype and flight units will be very similar externally to the preprototype units to be used on T-1 and S-2.

Construction of the test modification units for T-1 and S-2 has been completed. These units were delivered in February, about a month behind the originally scheduled delivery date. This delay has not affected completion of the T-1 vehicle. The late delivery was the result of machining difficulties encountered by a National Water Lift subcontractor that had limited experience with titanium. The use of this subcontractor was an effort to meet the early T-1 delivery schedule and is not planned for prototype or flight units. Difficulty was encountered during the deep hole drilling of the cylinder which resulted in the wall of the cylinder developing a hairline crack. The failure analysis report indicates that the material had the correct composition, no inclusions, and no residual stress concentrations, so that the fault in the unit is attributable only to the machining operation. The primary parts of the test modification prototype unit are shown in Figure 105.

A design review of the prototype shock absorber design was held at Kalamazoo on January 24, 1962. The

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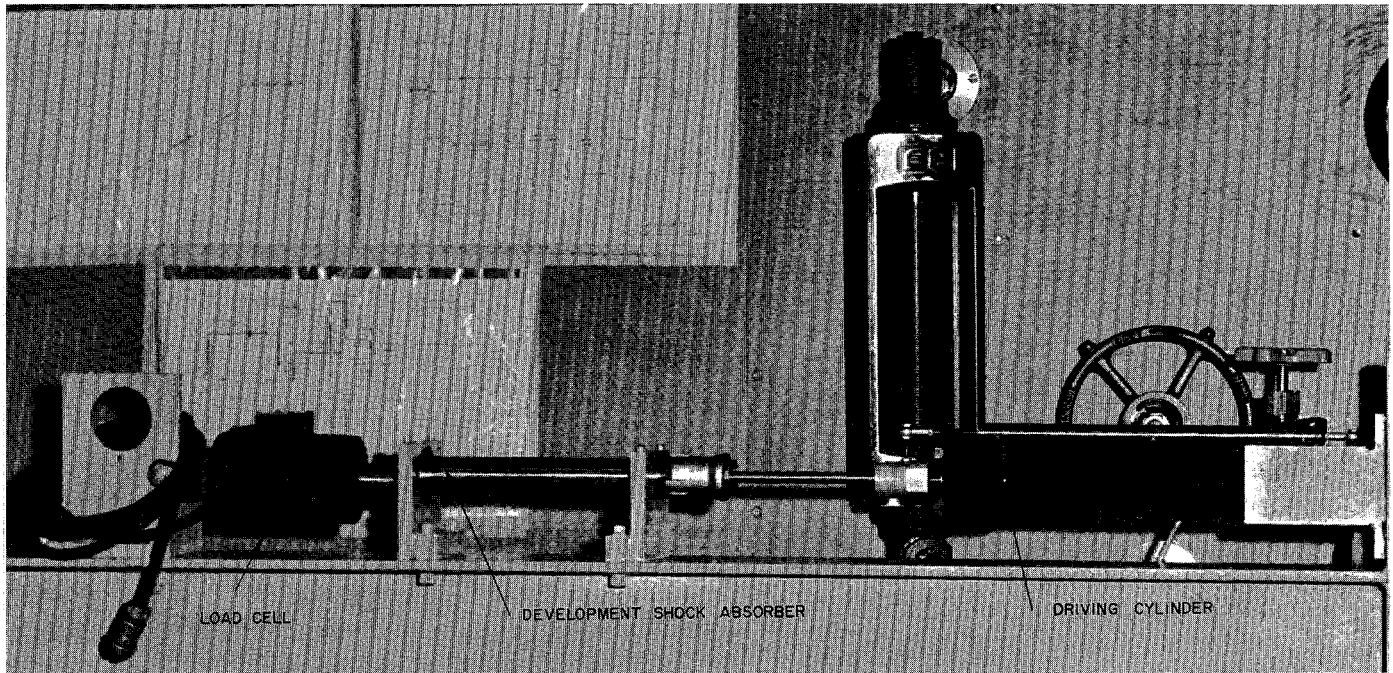


Figure 104. Shock absorber test setup

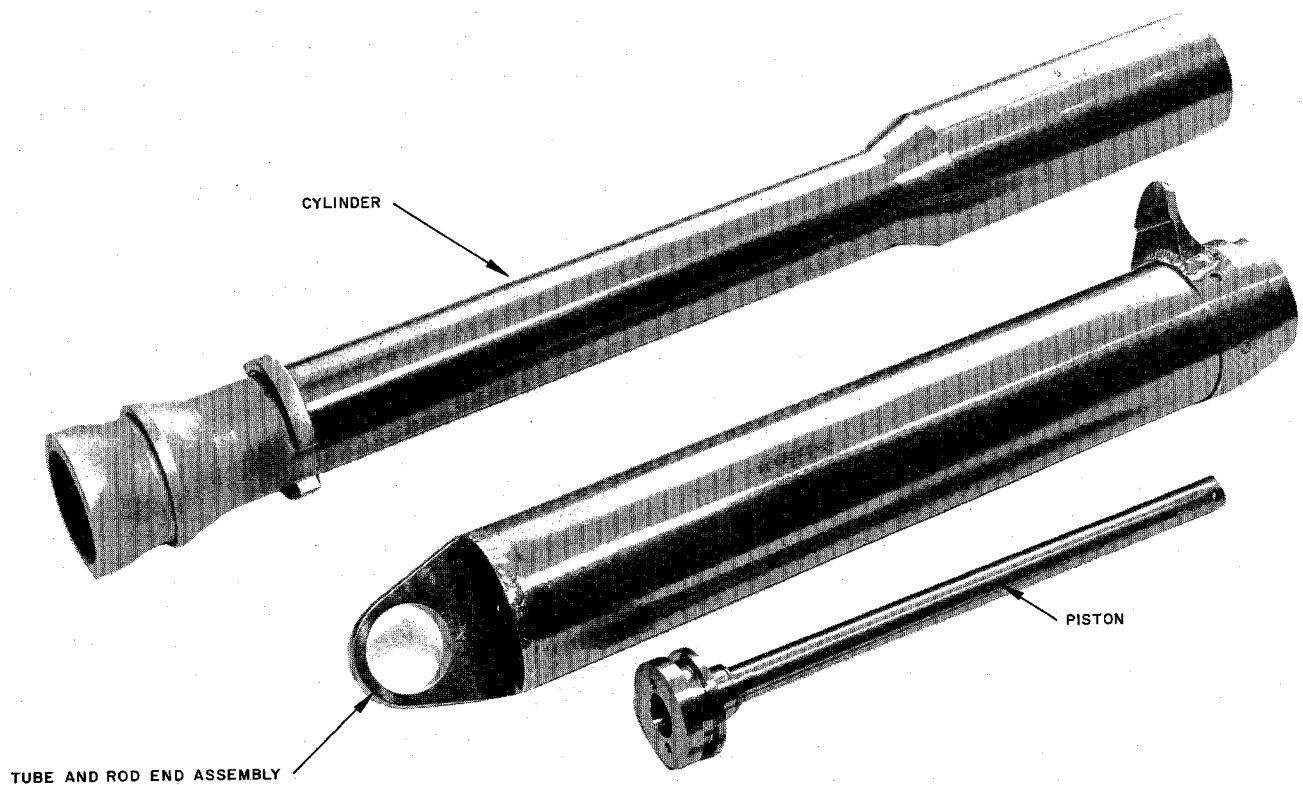


Figure 105. Components of test modification prototype shock absorber

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current design is shown in Figure 105. No significant faults were noted in the design although some minor improvements were suggested.

4. Structural Criteria and Loads Studies

Analog computer studies of the dynamic response of the spacecraft are nearing completion by Computer Engineering Associates (CEA). The first part of these studies was a static stress analysis, to be compared with the matrix analysis currently being performed by Hughes. The results of this analysis are still in terms of electrical analogs but will be reduced to mechanical terms after completion of the entire study program.

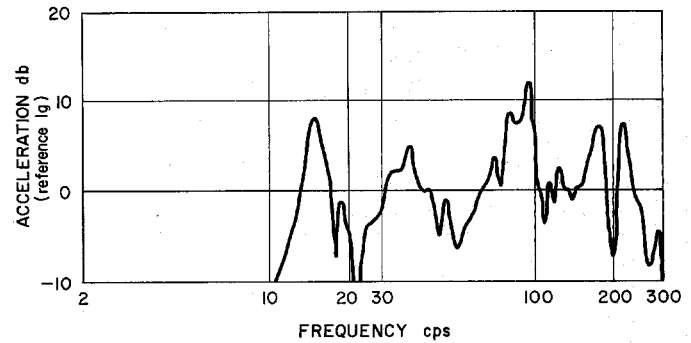
The dynamic behavior studies of the spacecraft during boost vibrations have also been completed. Although the resulting data have, to date, been reviewed only superficially, it has been possible to establish some general principles which will be reflected in revisions to the Structural Design Criteria Document. One observation is that internal structural loads due to resonances above 100 cps are smaller than those due to lower frequency resonances. This behavior is illustrated in Figure 106, a, b, and c, which show, respectively, the dynamic response of the subsurface sampler in two directions and the bending moment in the spaceframe member to which the subsurface sampler is attached. The bending moment at the high frequencies is low compared to the acceleration response whereas, at low frequencies, the bending moment is high compared to the acceleration response. CEA has also been studying the dynamic response of the spacecraft to the transient loads produced during lunar landing. These studies, the last in the program, have been completed and the results will be delivered by mid-February.

5. Vibration Tests

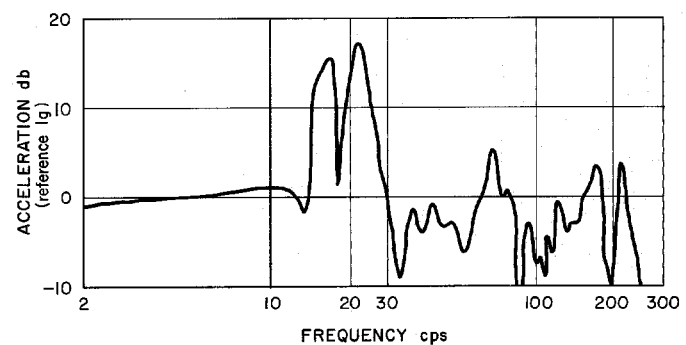
The strain gage data from the S-1 lateral vibration tests have been reduced. The input vibration level for these tests was $\frac{1}{4}$ g rms, corresponding to about $\frac{1}{3}$ the JPL Specification 30240A requirement.

The test results have been extrapolated to yield the stresses in the S-1 spaceframe which would result if full JPL Specification 30240A sinusoidal vibration levels were attained at the separation plane. The spacecraft was rigidly attached to the shaker table which acted as the separation plane in this study. The results are presented in Table 20.

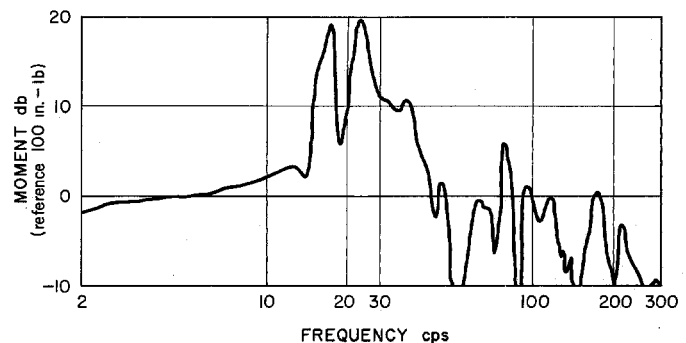
Figure 107 shows which instrumented spaceframe members are subjected to axial and bending stresses



a) Acceleration in Z-direction



b) Acceleration in Y-direction



c) Bending moment in tube to spaceframe

Figure 106. Calculated response of subsurface sampler — 1 g sinusoidal input at separation plane in Y direction

above 7000 lb/in². Large stresses occur in those members connecting the antenna to the spaceframe and in certain members of leg No. 1. In these members, however, the maximum axial and bending stresses do not occur for the same frequency or direction of the input vibration.

The retro-rocket lateral response is significantly different when the spacecraft is mounted on the interconnect than when it is mounted directly on the shake table. A comparison of Figures 108 and 109 indicates that the resonant frequency of the retro-rocket mounted on the interconnect is lower than when it is mounted

Table 20. Maximum stresses in S-1 spaceframe for lateral sinusoidal vibration at levels of JPL Specification 30240A (preliminary data)

Strain gage number ^a	Stress, lb/in ²	Frequency range on which maximum stress occurs, cps	Direction ^b of input vibration
9-1A	1420	40-50	A-3
9-8A	1160	20-25	A-5
9-10A	3140	20-25	A-1
9-12A	5380	20-25	A-1
9-15A	4160	75-100	A-2
9-17A	3780	75-100	A-2
9-1BZ	4720	40-50	A-5
9-8BZ	1750	50-75	A-4
9-10BZ	3540	20-25	A-1
9-12BZ ^c	5280	20-25	A-1
9-15BZ	4030	20-25	A-1
9-17BZ	2160	50-75	A-1
9-1BN	1850	20-25	A-5
9-8BN	2320	20-25	A-1
9-10BN	10,000	40-50	A-6
9-12BN	1900	20-25	A-1
9-15BN	1150	20-25	A-3
9-17BN ^c	1070	20-25	A-1
18-BX	24,700	8-9	A-5
18-BY	23,000	7-8	A-1
18-T	6250	7-8	A-1
3-15A	3600	20-25	A-1
3-15BN	950	50-75	A-3
3-16A	4000	50-75	A-6
3-16BN	1904	75-100	A-2
11-10BZ	3870	75-100	A-6
11-10BN	6140	50-75	A-3
15-10A	8650	20-25	A-1
15-10B(Y+30°)	8180	40-50	A-5
15-10B(X+30°)	8480	40-50	A-5
10-15B(Y+30°)	6420	20-25	A-5
10-15B(X+30°)	5680	40-50	A-2
11-12BZ	15,700	8-9	A-1
11-12BN	18,200	8-9	A-1
13-10A	7350	7-8	A-6
13-11A	11,750	8-9	A-1
13-12A	7100	9-10	A-3
14-10A	13,650	7-8	A-1
14-11A	5550	8-9	A-1
14-12A	3200	9-10	A-3
14-10BZ	3070	9-10	A-3
14-11BZ	2650	9-10	A-2
14-12BZ ^c	2520	8-9	A-6
14-10BN ^c	7300	9-10	A-2
14-11BN	8320	7-8	A-4
14-12BN	7440	7-8	A-1
10-11A	2360	40-50	A-3
10-11BZ ^c	4680	20-25	A-1
10-11BN	6600	20-25	A-3

^a See Figure 107 for location and axis notation. 9-1A means axial stress in member connecting joints nine and one. Stress is measured near joint nine. 9-1BZ means bending stress in Z direction, etc. Gage number 18 is located 2 in. above antenna ring.

^b See Figure 107 for direction of input vibration.

^c Instrumentation lacking for some orientations.

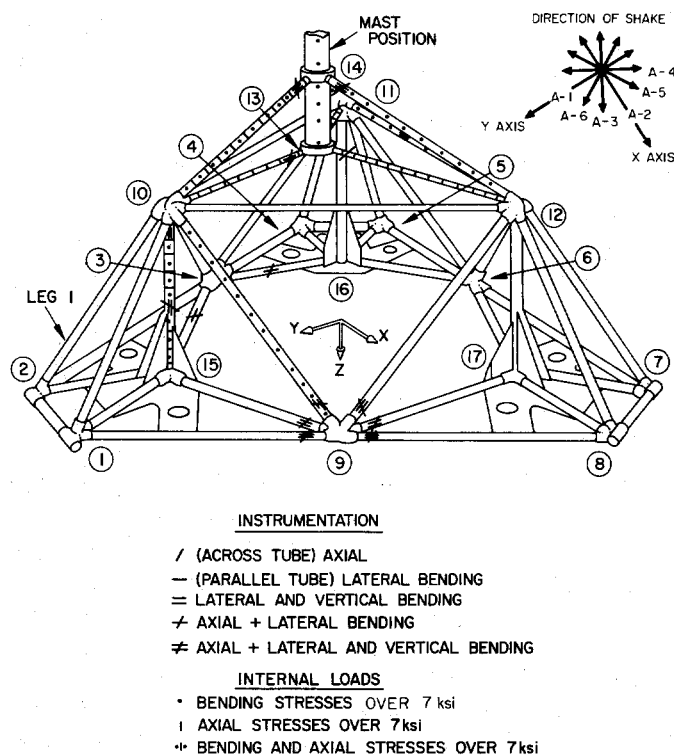


Figure 107. Instrumentation of S-1 for vibration tests

on the shake table. This is probably due to local flexibility of the interconnect which decreases the retro mounting spring stiffness. Retro-rocket response amplitude is also lower when the spacecraft is mounted on the interconnect. For both mounting cases the retro-rocket acceleration exceeds the values of JPL Specification 30240A. In type approval testing on S-2 and S-3, reduction in the input vibration levels will be utilized to keep the retro-rocket center of gravity acceleration below 2 g.

A firm date has not yet been established for the acoustic tests planned at Rocketdyne. This date will be influenced by the availability of the S-1 vehicle and the firing schedule at Rocketdyne. The test will be run with the original S-1 dummy units. However, in order to obtain engineering information relative to the behavior of thin plates in acoustic fields (since compartment walls are 0.008 to 0.012 in. thick), plans are being developed to mount a number of thin plates on the test vehicle. These plates would be instrumented with strain gages so that time histories of the plate stresses could be recorded and analyzed. The amount of instrumentation planned for the test has been increased from 12 channels to 30 channels, comprised of 12 accelerometer channels and 18 strain gage channels.

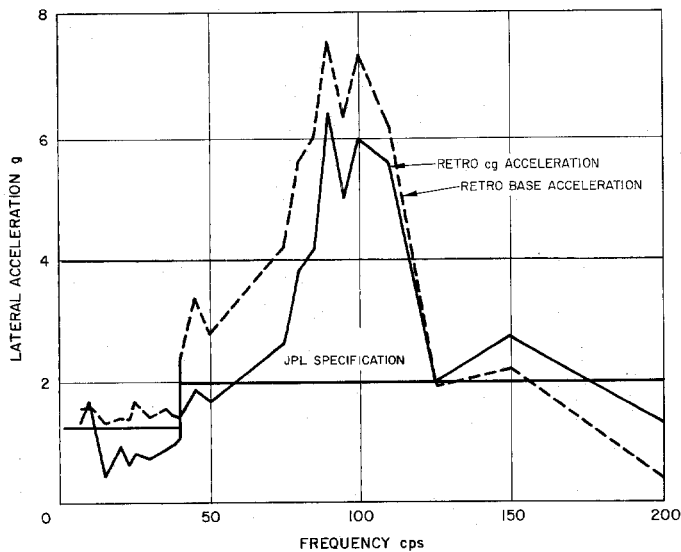


Figure 108. Dummy retro-engine response—S-1 mounted directly on vibration table and vibrated in Y-direction

A type approval vibration test has been developed for the vernier system mounted on the S-5 spaceframe which is intended to simulate the input to the system arising when all the components are present on the frame and the frame is vibrated at type approval levels proportional to the JPL Specification 30240A levels. The sinusoidal levels are shown in Figure 110. It was assumed that Compartments A and B, the subsurface sampler motor, and the soil analysis group would be the only elements that affect the vernier system through the spaceframe. The JPL specification level was augmented at the estimated resonant frequencies of these elements in proportion to their response in the S-1 lateral vibration test and their distances from vernier system components. No part of the system need see more than 12 times the depressed JPL vibration specification, which is also shown in Figure 110. Random vibration levels were estimated to be 6 g

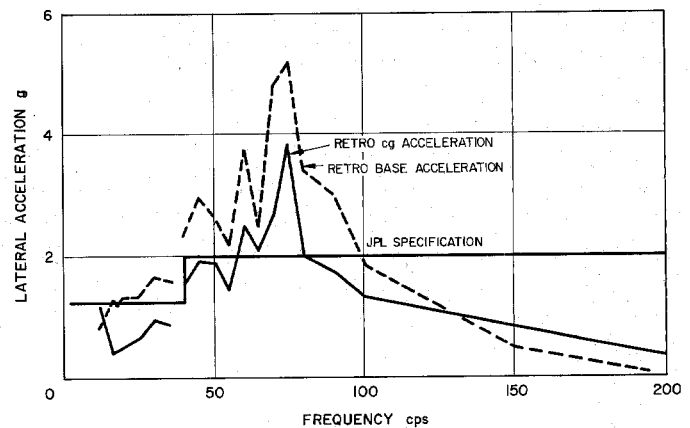


Figure 109. Dummy retro-engine response—S-1 mounted on interconnect and vibrated in Y-direction

rms for 2 min and 3 g rms for 10 min. The random vibration is white Gaussian acceleration from 100 to 1500 cps. Partial verification of these levels as adequate type approval levels will be made during S-2 or S-3 vibration tests.

Retro-rocket and vernier motor vibration inputs to the structure are expected to be low. It is felt that vibration inputs from the verniers would not exceed ± 20 lb, which is not significant in design of the spaceframe structure. The vibration forces from the retro-rocket could be substantially greater and hence warrant measurement, if only a confirmation-type check.

As a result, plans are being prepared to instrument a test stand at the Thiokol Elkton facility and measure nine accelerations (three at each retro-rocket attach point) on one, to possibly four, flight weight retro-rockets beginning in late March 1962. Simplified theoretical estimates of the spacecraft, test stand, and retro-rocket impedances would be made concurrently to determine whether a structural design problem exists. If it does, or there is

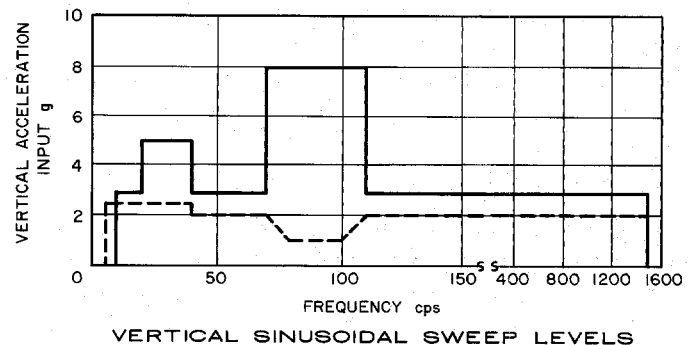
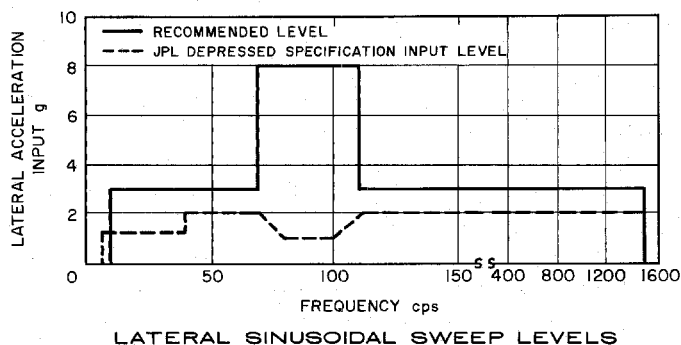


Figure 110. Vibration test levels for vernier propulsion system

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uncertainty that it does, then accurate impedance measurements of the test stand, spacecraft, and retro rocket would be made to ascertain the expected retro rocket vibration input to the spacecraft. The phasing of the impedance measurement equipment purchase is presently under consideration. If the equipment is purchased and checked out now and then a problem appears, quantitative data could be obtained by October 1, 1962. If equipment is not purchased until it is certain that a problem exists, quantitative data would not be available until February 1, 1963.

6. Flight Vehicle Instrumentation

A procurement specification for the flight vehicle accelerometer-amplifier systems is in preparation and is expected to be completed by February 15, 1962. Seven systems are to be installed on each flight vehicle. The purpose of the accelerometer-amplifier systems is to sense mechanical vibration during boost, retro and vernier vibrations during descent, and mechanical shock during landing. The accelerometers will be located near the bases of the spaceframe vertical columns to sense the vibration inputs from the *Atlas-Centaur* through the interconnect mounts, the retro-rocket vibration through its supports which tie in below the accelerometer installation, and the landing shock arising principally from the crushable structure. The accelerometers will be oriented to provide the 6 deg of freedom rigid body acceleration input during touchdown.

The amplifiers are to be mounted as a single unit on Landing Leg 3. This location was chosen because of weight and balance considerations and to expose the amplifiers to solar radiation for thermal control.

The accelerometer and amplifier will be hermetically sealed. The accelerometers will be piezoelectric crystal units with natural frequencies greater than 35,000 cps. The amplifiers will be miniature, transistorized devices, with signal conditioning (filtering above 1000 cps, limiting to ± 2.5 v and a second biased output for the tape recorder). Each accelerometer-coax-amplifier system will produce a full-scale output of ± 2.5 v for an acceleration input of ± 40 g over the frequency range of 5 to 1000 cps. Each system is expected to weigh less than 0.5 lb.

The accelerometer-amplifier systems will operate over the temperature range -40°F (-100°F for the accelerometer) to 185°F and will not be degraded by exposure to the heat sterilization temperature (257°F). The systems will also operate properly when exposed to a vibration environment of ± 30 g or 100 g sawtooth shocks.

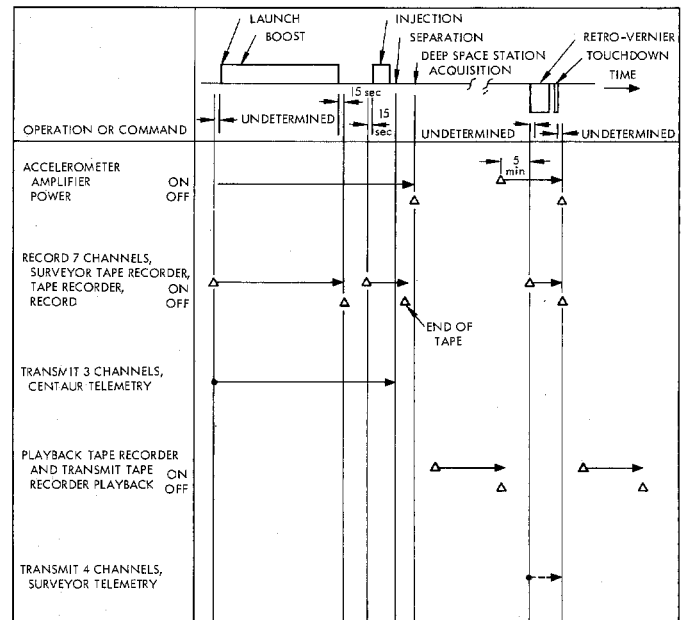


Figure 111. Engineering data sequencing

The amplifier will be tested for electrical operation during shock and vibration input. The best estimate of the data acquisition sequence is presented in Figure 111.

7. Structural Review of Scientific Instruments

The structural design of the gas chromatograph, seismograph, the X-ray diffractometer, X-ray spectrometer, and surface and subsurface source instruments were reviewed during visits to the respective subcontractors. In general, insufficient stress analysis was produced to ascertain the capability of the instruments to successfully withstand the *Surveyor* load and vibration environment.

For gas chromatograph and seismograph, the design was based on the vibration and shock levels of Hughes Specification X224810 which has lower levels than the current Hughes Specification 224810B. Damage due to acoustics had not been fully considered and shipping containers had not been designed for these instruments. The seismograph subcontractor felt that the transportation drop test in Hughes Specification X224810 should be removed because it is too severe a requirement. The seismograph will be tested to the shock and sinusoidal vibration levels of the specification, but the random tests will be omitted. The baseplate of the gas chromatograph with dummy weights attached was vibration tested to a 5 g input. Transmissibilities as high as 25 were found.

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The stress analysis calculations for the other scientific instruments were based on design load factors of 30 g longitudinally or 15 g laterally. Weight reduction possibilities in the X-ray diffractometer and X-ray spectrometer could be confirmed by more detailed stress analysis.

It is felt that further study of the total environment presented in Hughes Specification 224810B should be made by the subcontractors to obtain comprehensive design load and thermal information and that these levels should be used for test.

8. Spaceframe Stress Analysis

A matrix loads analysis of the spaceframe has been completed for the vertical boost load factors given in the Hughes Structural Design Criteria Report. In the matrix loads analysis, the spaceframe was assumed to be mounted on three sets of springs in order to simulate the flexibility of the adapter. Eccentricities present in the clusters at the truss joints have been included in the analysis.

The results obtained by applying 1 g in the vertical direction to all components attached to the spaceframe for the S-2 configuration are shown in Table 21. Axial loads, torques, and bending movements acting at the ends of some of the more highly loaded tubes are given. The spaceframe numbering system and sign convention are shown in Figure 112. An internal load distribution for unit sinusoidal load factors acting in the vertical direction has been obtained and must be superimposed on the quasi-static loads (which are obtained from the 1 g case) in order to obtain the total vertical boost case. By means of similar procedure, the total lateral boost case will be obtained by applying load factors in several lateral directions. Margins of safety will then be calculated for the maximum combined loading cases occurring during boost.

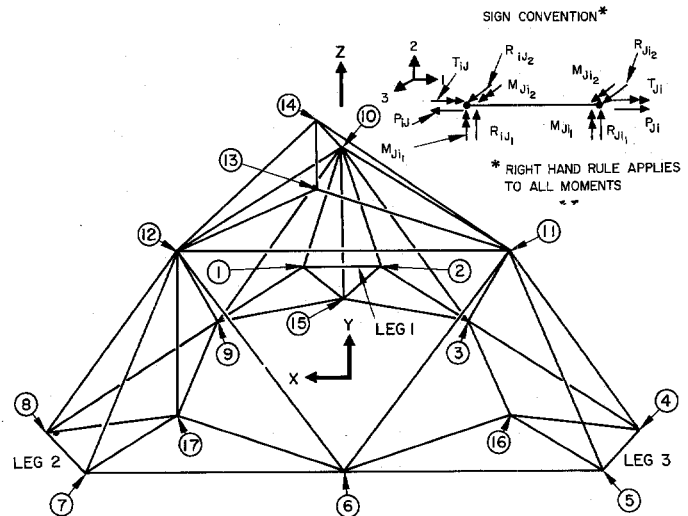


Figure 112. Spaceframe numbering and sign conventions

A fatigue analysis is being performed by considering oscillatory loads during boost and descent. The complete cyclic load spectrum during the boost phase will then be obtained by considering 11 different cases for each given load direction. These cases differ from each other in that the phasing of the ten heaviest components is successively changed. To the boost spectrum will be added 9000 cycles of oscillatory load during the descent phase.

For static simulation of a touchdown condition, a similar analysis will be employed, with the exception that external reactions will be applied to the spaceframe through the landing gear members acting in combination with the crushable honeycomb blocks. The S-2 analysis is expected to be complete by the end of February.

9. Static Load Test (S-1)

The static load tests of Model S-1 were completed on January 26, 1962. The tests were designed to supplement

Table 21. Axial loads, bending moments, and torques for critical S-2 spaceframe elements
($n_z = 1.0$)

Tube		P_{1J}	P_{J1}	T_{1J}	T_{J1}	M_{1J1}	M_{J11}	M_{1J2}	M_{J12}
i	J	lb	lb	in.-lb	in.-lb	in.-lb	in.-lb	in.-lb	in.-lb
11	10	29.0	32.0	- 3.4	3.4	7.5	- 37.0	- 58.0	- 20.0
3	11	-110.0	-121.0	12.0	-12.0	45.0	15.0	- 78.0	- 91.0
1	9	- 9.5	- 10.1	85.0	-85.0	- 28.0	- 26.0	25.0	- 0.7
4	5	7.2	7.2	2.5	- 2.5	-143.0	104.0	62.0	- 30.0
15	9	107.0	107.0	85.0	-85.0	206.0	180.0	- 1.2	- 4.3
15	10	209.0	208.0	- 2.2	2.2	9.1	10.4	740.0	297.0
15	2	21.0	21.0	-12.0	12.0	-506.0	-172.0	- 9.2	- 22.0
8	12	6.9	6.4	3.6	- 3.6	- 74.0	- 59.0	27.0	3.7

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the lateral sinusoidal vibration test, to substantiate assumptions used in the matrix loads analysis, and to verify stress calculations. The magnitude of applied loads was in all cases only large enough to establish accurately load-deflection curves and obtain strain gage readings. Ultimate loads could not be applied because the S-1 spaceframe, which was fabricated in mid-1961, was constructed of relatively low strength aluminum. Although 100% strength was not obtained, S-1 is a very close simulation of the present spaceframe in both configuration and elasticity.

Spring constants were obtained for the interconnect which was designed and built at Hughes for use in S-1 vibration and static tests. Loads were applied at the three interconnect-to-spaceframe attach points and absolute deflections were measured at each point. Strain gage readings were also taken on the interconnect shell. The test setup for the interconnect is shown in Figure 113.

With the spaceframe mounted on the interconnect, loads were applied along the principal axes through the center of gravity of the following simulated components: retro-rocket, Compartment A, Compartment B, soil analysis group, flight control sensor group, subsurface sampler, and antenna. Displacements of each component and spaceframe attach point were taken in order to measure the elasticity of support structure and spaceframe. Strain gage readings were taken on the spaceframe at the locations shown in Figure 85 of Bimonthly Progress Summary SS-103. Additional spaceframe spring constants

were obtained by applying lateral loads in the plane of the upper spaceframe tubes. Figure 113 shows the S-1 model spaceframe and simulated components (with the exception of the antenna and solar panel) mounted on the interconnect in the boost configuration.

The static test report for S-1 will be issued by the end of February. The interaction of the largest loads on the spaceframe for the boost conditions in terms of stresses and displacements can be determined by superposition of portions of the test data obtained. A comparison of spring constants for the Hughes and GDA interconnect structures will be made when the GDA interconnect is tested in conjunction with Model S-2.

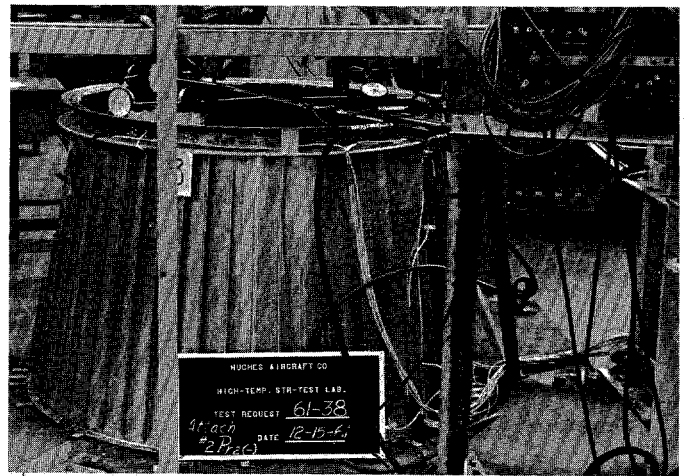


Figure 113. Interconnect load test setup

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I. Propulsion

1. Introduction

The spacecraft midcourse correction, attitude control during main retro operation, and incremental velocity corrections are accomplished by a liquid bipropellant vernier propulsion system. Main spacecraft deceleration during the terminal phase of flight is performed by a spherical solid propellant retro-rocket engine. Development of the vernier system and the main retro-rocket engine has been subcontracted to the Reaction Motors Division (RMD) and the Elkton Division, respectively, of Thiokol Chemical Corporation.

2. Progress Summary

Fabrication of the initial flight-weight components of the main retro-rocket has been completed, and the case assembly passed the hydrostatic proof pressure test. Tests conducted to date, using heavy-wall components, have identified the major development problems as propellant cracking and sterilization.

The propellant formulation has been changed from the HA series PBAA to the HC series to eliminate low-temperature propellant cracking. The new propellant, however, is not able to withstand thermal sterilization, and alternate sterilization techniques are being studied. The initial test firing of a sterilized *Surveyor* main retro-rocket engine, loaded with the original propellant (HA

series), resulted in the engine blowing up after 120 ms burning. The exact cause for this failure has not yet been identified.

A comprehensive design study for a retro-rocket engine for use in a 2100 lb spacecraft has been completed. Performance parameters and the detail configuration have been established. In Figure 114 and Table 22, the design and performance characteristics of the current and light-weight engine configurations are compared. The primary differences between the two are the amount of propellant and the grain configuration. The 2100 lb version has a reduced propellant web thickness, has seven small star points, and is cast to a reduced height. Both configurations use the same propellant formulation, case, case attachment, nozzle expansion cone, and pyrogen components. Detail components and tooling designs have been completed, and additional core billets have been purchased. The 2100 lb engine configuration could be available for the first scheduled spacecraft launch.

Initial testing of the complete vernier system has proved satisfactory. Additional tests of chamber durability have demonstrated a capability of 25 min of accumulated firing time. Start and shutdown transients are within design specifications. Vernier chamber performance has improved. Specific impulse limits are now met at low and medium thrust. At maximum thrust, specific impulse is below design requirements.

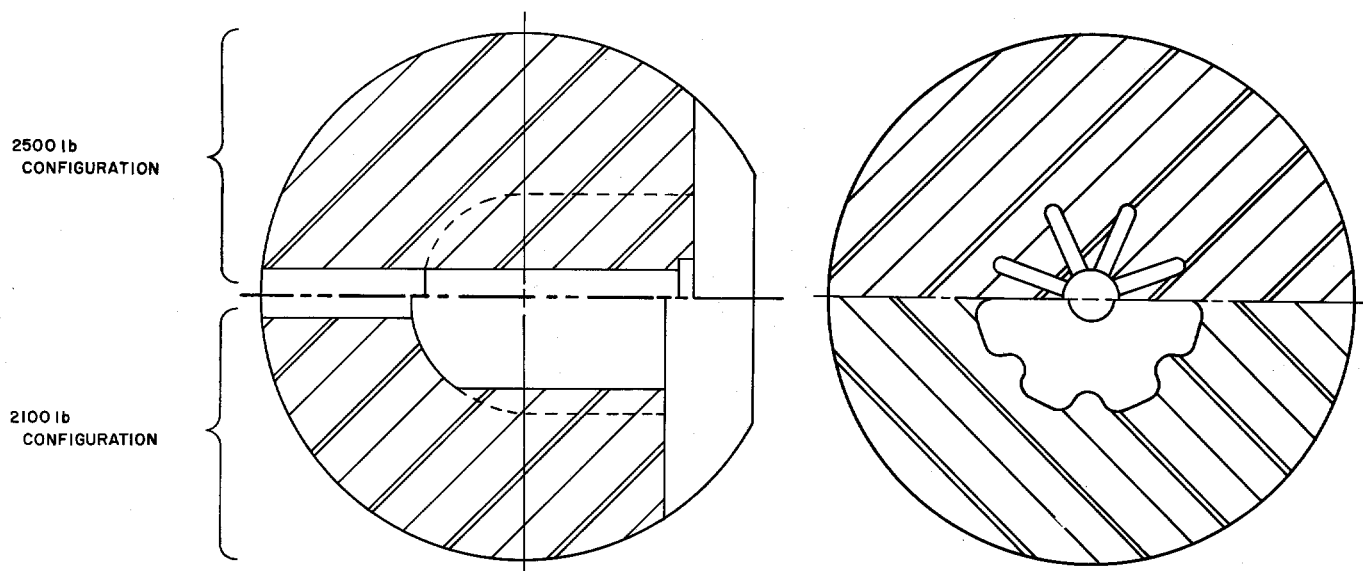


Figure 114. Grain configurations of current and lightweight engines

3. Main Retro-Rocket Engine

a. Rocket engine tests. Two rocket engines were tested at sea level conditions and an additional engine was cast during the report period. The test of Engine 3, primarily a burnout of a rejected engine, utilized a tungsten-molybdenum throat insert. Evaluation of the insert was not conclusive because of leakage at the interface between the insert and the graphite section of the nozzle, which caused severe erosion of the interface. The status of nozzle development is discussed later.

The test of Engine 4, which had been subjected to the heat sterilization environment, resulted in the engine blowing up after 120 ms operation. The pressure-time record for Engine 4 is shown in Figure 115. Thorough investigation of the failure has been initiated. Samples

of propellant recovered from the exploded engine are being analyzed for changes in physical and ballistic properties. Possible causes of the failure are: (1) sterilization-induced effects on the propellant, (2) propellant-to-liner bond failure, and (3) nozzle blockage caused by star-point failure. The evidence now available is insufficient to favor any one of these theories.

Liner TL-H-703 was used for the first time in Engine 4. No separation was visible in the prefiring visual and x-ray examination of the engine.

Engine 5 was rejected because of failure of the bond between the liner and propellant. The extensive separation experienced in this engine is attributed to improper application of an adhesive between the liner and propellant, as discussed later.

Table 22. Comparison of main retro-rocket engines for 2100 lb and 2500 lb spacecraft configurations

Characteristic	2100 lb spacecraft engine		2500 lb spacecraft engine
	50°F	70°F	70°F
Maximum thrust, lb	9345	9500	10000
Average thrust, lb	8170	8370	9260
Average pressure, psia	555	569	560
Total impulse, lb-sec	350000	350000	420100
Specific impulse, sec	294	294	294
Action time, sec	43	42	45.5
Propellant weight, lb	1190		1428

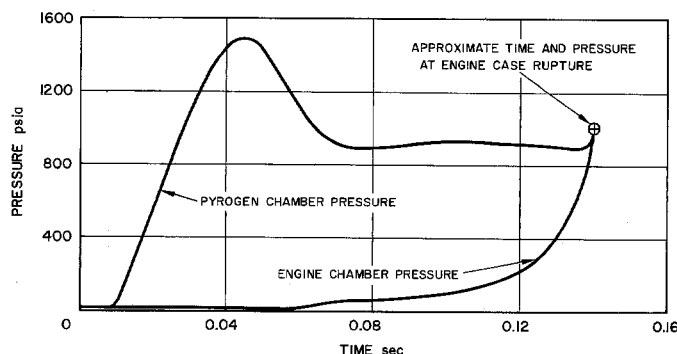


Figure 115. Malfunction of Retro-Engine 4

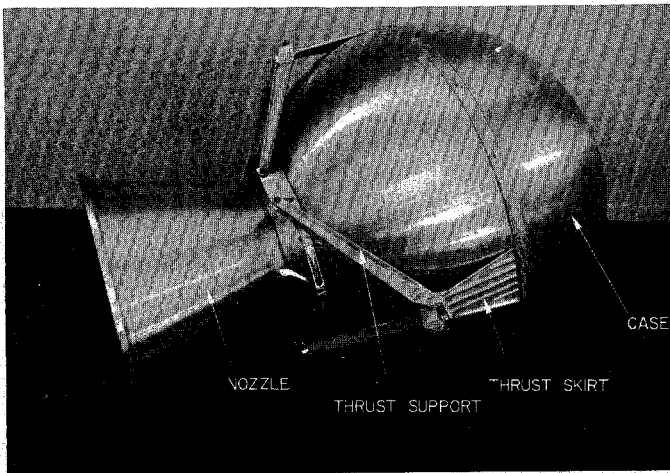


Figure 116. Mockup of retro-engine

b. Flight-weight case. Fabrication of the first two flight-weight cases has been completed, and hydrostatic proof testing of the initial case was successful. During this test the attachment fittings were subjected to an axial load equivalent to maximum engine thrust, while the case was pressurized. Minor rework of the attachment strut was necessitated by a localized interference of the strut and aft case hemisphere during case dilation. Redesign will provide adequate clearance on subsequent case assemblies. The first case will be used in the engine test program; the second case will be used as an inert engine which will be subjected to environmental testing with the spacecraft. The configuration of the flight-weight retro-rocket engine is shown in Figure 116.

c. Propellant development. Thermal analyses of the present main retro-rocket engine installation indicate that certain parts of the engine case will cool during transit to temperatures well below the minimum acceptable for the HA series propellant. A study was therefore made to determine the effects of various thermal control systems.

The results indicate that only an active temperature control system using excessive amounts of heating power can maintain the engine temperature above the lower temperature limit of the HA series propellant. The propellant formulation has therefore been changed to a type with improved low-temperature properties to eliminate the possibility of low-temperature cracking of the propellant. The propellant selected is a PBAA type, TP-H-3061. The propellant has superior physical properties to the TP-H-3053B previously used, as shown in Table 23. The propellants are identical except for fuel binder, physical properties, and ability to withstand heat sterilization. Minor burning rate adjustments have been made to the TP-H-3061 formulation to provide a propellant suitable for use with both engine configurations.

Results of investigations indicate that TP-H-3061 cannot be heat sterilized and that both TP-H-3061 and TP-H-3053B are not sterile as cast. At present, the technique for sterilization of the propellant has not been resolved. Investigations of sterilization by irradiation and combinations of irradiation and heat have been initiated.

d. Liner. A liner change was introduced on Engine 4 and subsequent engines. The liner, TL-H-703, exhibits physical properties more compatible to heat and irradiation sterilization methods and the thermal environment. A comparison of the physical properties is presented in Tables 24 and 25. The adhesive failure of the propellant to liner bond experienced in Engine 5 resulted from an uneven application of the amine cure solution to the partially cured liner. Further evaluation of liner materials has been initiated to ensure compatibility with the TP-H-3061 propellant.

e. Nozzle development. A satisfactory nozzle throat configuration has yet to be demonstrated. In Engine 1 and 2 firings, nozzle erosion was 20 to 25%. A low-density carbon had inadvertently been used in these firings. Engine 3 was fired with a tungsten-molybdenum throat insert, bonded to high density carbon. The bond between the carbon and the insert failed, and erosion in this area

Table 23. Comparison of physical properties of HA and HC series PBAA propellants

Characteristic	HA series (TP-H-3053B)						HC series (TP-H-3061)		
	Pre-sterilization			Post-sterilization			Pre-sterilization		
Temperature, °F	20	80	110	20	80	110	20	50	110
Stress, lb/in ²	110	87	72	120	81	58	123	105	92
Strain, in./in.	0.202	0.201	0.193	0.135	0.138	0.107	0.428	0.417	0.422
Modulus, lb/in ²	873	632	515	1230	751	686	1029	631	417
Tear, lb/in.	14	11	8	15	12	8	27	21	17

Table 24. Comparison of tensile properties of TL-H-700 and TL-H-703 liners

Test temperature, °F	TL-H-700		TL-H-703	
	Stress, lb/in ²	Elongation, %	Stress, lb/in ²	Elongation, %
140	132	50	186	120
100	226	40	250	160
60	233	40	218	145
20	283	50	276	160

Table 25. Comparison of typical metal-to-liner shear strengths of TL-H-700 and TL-H-703 liners

	Test temperature	Stress, lb/in ²	Type of failure
TL-H-700	Ambient	175	Adhesive
TL-H-703	Ambient	350	Adhesive

was excessive. For the Engine 3A firing, pyrolytic graphite annuli were placed in the throat section of the nozzle. The annuli were blown out of the nozzle 28 sec after ignition. The pressure trace of the firing, shown in Figure 117, indicated there had been no appreciable nozzle erosion.

Thiokol believes a monolithic, high-density carbon nozzle can be used in the retro engine with a maximum erosion of 15%. The next five firings are tentatively scheduled with this configuration. If erosion is excessive, one of the composite nozzles (pyrolytic graphite or tungsten-molybdenum) will be used. A tungsten-molybdenum insert would increase the inert weight approximately 3 lb. Thiokol is pursuing the monolithic graphite approach because of the simplicity and inherent reliability of that configuration.

f. Altitude test status. Installation of the altitude test facility at Elkton has been completed and the facility is operational. It will permit retro-rocket engine testing at a simulated altitude of 70,000 ft by the diffusion pumping action of the rocket exhaust.

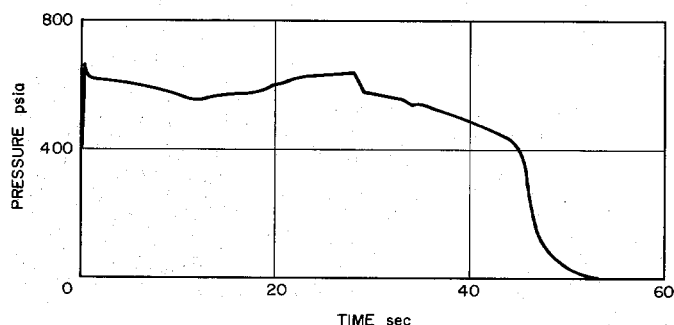


Figure 117. Performance of Retro-Engine 3A

Initial testing of retro-rocket engines at Arnold Engineering Development Center (AEDC) has been rescheduled for March 26, 1962, to permit evaluation of the propellant change under simulated altitude conditions. The tests at AEDC permit accurate determination of tail-off characteristics at a simulated altitude in excess of 100,000 feet, because of the exhaust pumping capacity over and above the jet diffuser action.

g. Thermal control. A thermal gradient analysis involving various combinations of engine support materials, passive thermal control coatings, and insulation materials has been conducted on the IBM 7090 computer. The results of this study show that the most effective means of passive thermal control is a jacket composed of approximately 25 to 30 layers of aluminum-coated mylar on the engine case and 50 to 60 layers on the nozzle extension. The suitability of this approach for installation in the spacecraft is being checked.

4. Vernier Propulsion System

The vernier propulsion system consists of a helium pressurization system, liquid propellant tankage system, and three thrust chamber assemblies, each of which can supply any thrust level between 30 and 104 lb on command. Monomethyl hydrazine is used as the fuel as well as for regenerative cooling of the thrust chamber. Nitrogen tetroxide with 10% nitric oxide is used as the oxidizer. This system is capable of providing approximately 50,000 lb-sec total impulse.

The program has been rescheduled during the report period, with the development program now expected to be completed on April 15, 1962 instead of March 15, 1962. Additional rescheduling of the fabrication and acceptance testing phases has permitted retention of the originally scheduled delivery dates for flight and prototype systems.

The major accomplishment of the report period was the initiation of complete vernier propulsion system testing. The following paragraphs present details of the more significant items of accomplishment.

a. Complete vernier system testing. The first partial vernier system firing was accomplished on December 30, 1961 in the altitude test facility. This firing was conducted on a system which included the helium supply system, a two-tank propellant feed system, and a single thrust chamber assembly. The test marked the first time the characteristics of the thrust chamber and flight feed system were checked for operational compatibility. Test results indicated no adverse influence between the two. Data obtained during the limited throttling operation from this testing indicate excellent compatibility of the system components. The first complete vernier system test was accomplished late in the report period after delays caused by minor component problems, discussed in later paragraphs. The complete system contains six propellant tank assemblies and three thrust chamber assemblies (Fig 118). System testing is continuing and is also being carried out in the altitude test facility at

RMD. Data from complete system tests are not yet available.

b. Thrust chamber development. The durability of the thrust chamber has been established with the accumulation of approximately 25 min firing time and seven starts on a single thrust chamber. Approximately half the time accumulated was under maximum thrust conditions, and approximately six min were under transient throttling conditions.

Thrust chamber performance development is continuing in an effort to improve the specific impulse at maximum thrust. The performance at minimum and at mid-range thrust is now somewhat higher than specified, thus indicating improvement in this area over previously tested configurations. Vernier chamber performance characteristics are shown in Figure 119.

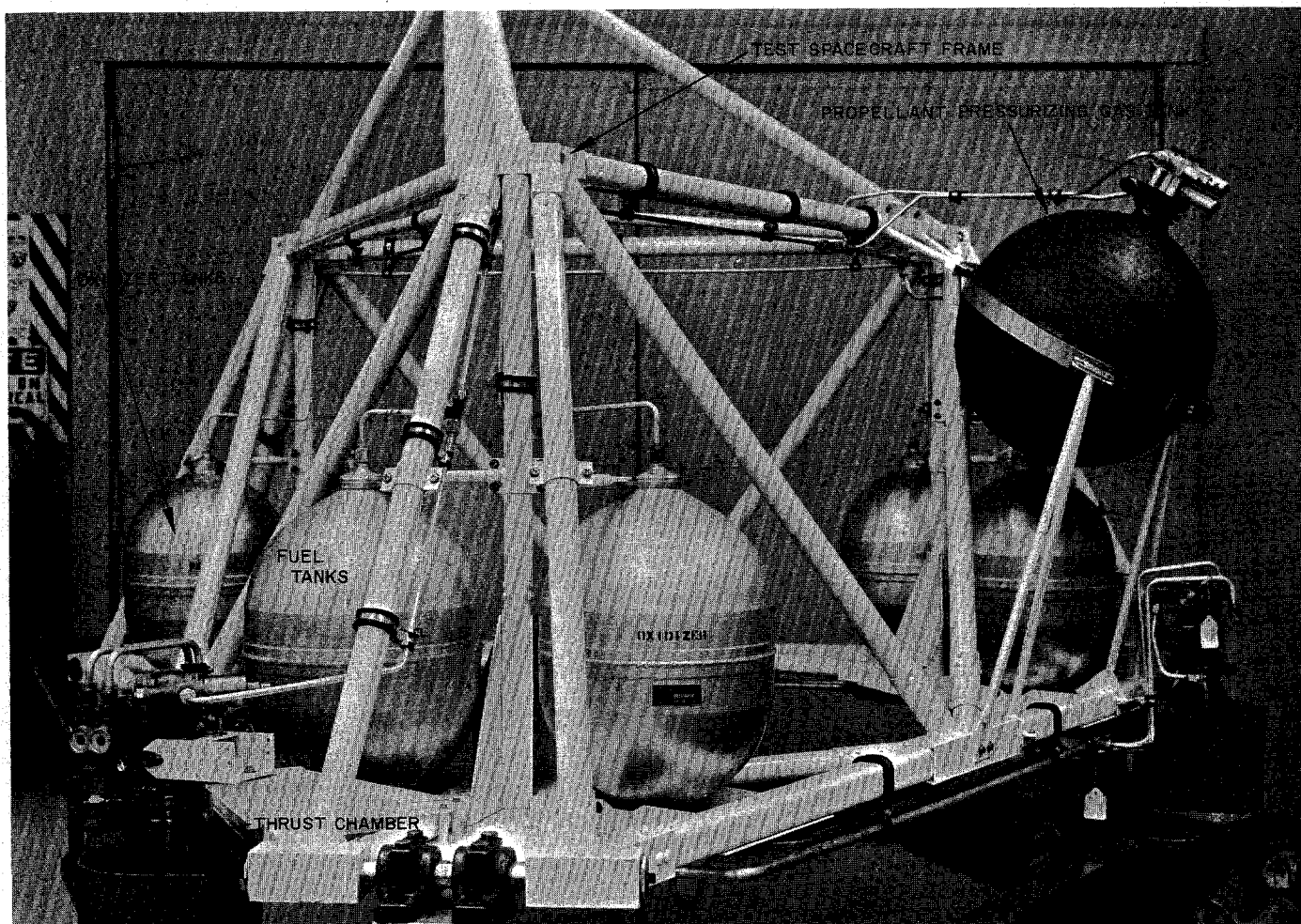


Figure 118. Vernier propulsion installation on test spacecraft

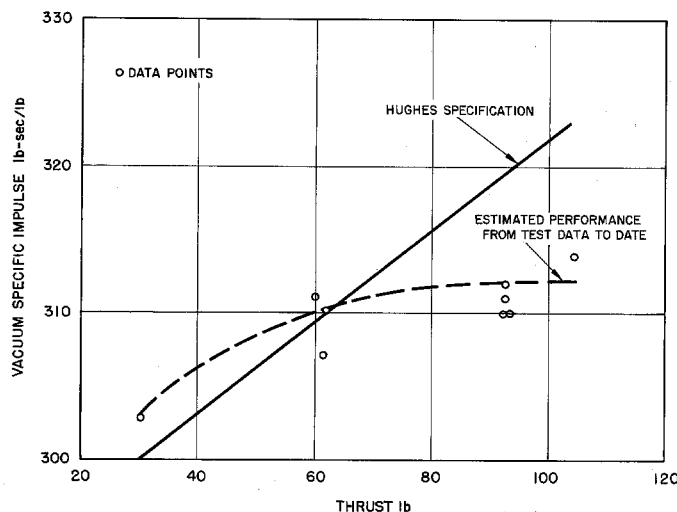


Figure 119. Performance of vernier thrust chamber

Early testing of the flight thrust chamber assembly (including the thrust chamber, on-off valve, and throttle valve) indicated that the required start-and-shutdown transient times were not being achieved. The injector volume has now been reduced and the required transient response has been demonstrated.

Minor thrust chamber assembly problems were encountered in injector brazing. Evaluation of two solutions to these problems — a furnace rather than induction brazing procedure and a bolted assembly requiring no brazing — is near completion.

c. Component development. The following paragraphs present the status of each component.

- (1) Helium tank — Type approval testing will be completed in the very early part of the next report period.
- (2) Helium fill-release-dump valve — Completion of type approval testing has been delayed until the next report period because of an O-ring leakage problem which has been recently solved. This valve, which is directly connected to the helium tank, is used to fill the pressurant system, to release the pressurant to the propellant storage system, and to dump the pressurant overboard in case of emergency.
- (3) Helium-regulator-check valve-relief valve package — Development testing has indicated that satis-

factory regulation is obtained with the bellows-type second stage regulator, a condition which did not exist with the earlier O-ring type system. Type approval tests will be initiated early in the next report period.

- (4) Propellant tanks — Fabrication problems have been solved and tankage is being supplied to the program on a routine basis.
- (5) Propellant expulsion bladders — Testing has resulted in a demonstration of 20 collapsing cycles on a single bladder with no leakage detected. However, several bladders delivered for complete system tests have been found to have leaks prior to any actuation. This problem is being investigated and solutions are being explored.
- (6) Propellant throttle valve — The throttle valve controls propellant flow by means of a pilot valve having two reeds which vary effective flow areas and are actuated by a single torquemotor. The effects of pressure variation are eliminated through the use of the spool valves. Minor sticking problems have been solved by changing the coating on the spool valves in the pressure-balancing section of the valve. Completion of the type approval testing is expected to occur approximately midway through the next report period.
- (7) Propellant on-off valve — Type approval testing is expected to be completed midway through the next report period. This valve controls both propellants through the use of solenoid-piloted, helium-actuated poppets.

d. Thermal control. All vernier components for Sector III of the MT-1 thermal mockup have been fabricated and are presently being fitted with the appropriate thermal control coatings and/or heaters. Contact has been made with potential spacecraft thermal control component vendors, and planning purpose quotations have been requested. It is anticipated that the capabilities of these facilities will be evaluated through their respective efforts in preparing MT-1 components for testing. Means of providing thermal insulation at the support joints of the engines, fuel and oxidizer tanks, and helium tank have been established for incorporation in the flight design.

J. Spacecraft Vehicle and Mechanism for Basic Bus

1. Spacecraft Vehicle

a. Configuration. The spacecraft configuration was put under design control in December 1961. Since that time a group of small controlled changes have been made to correct a lateral center of gravity shift that developed due to weight changes in various subsystems. The significant changes incorporated since the last report are listed below and are shown in Figure 1.

- (1) The attitude control jet pressurization tank has been relocated to a position inboard of TV Camera 4. A common attach structure can be used to integrate this tank, Cameras 3 and 4, and the thermal diffusivity experiment to the spaceframe.
- (2) The signal data converter (for the doppler/altimeter radar) was moved from the region of Leg 1 to a similar location near Leg 3 but in the same sector.
- (3) The klystron power supply and modulator (doppler/altimeter radar) has been shifted outward radially and its attach structure has been modified to permit a common attach point at the spaceframe with the lower support of Omni-directional Antenna No. 1.

Major effort is now being directed to the detailed design of the subsystem attach structure, thermal insulation, and alignment adjustment methods. To aid in this design effort, a full-scale engineering mockup vehicle

(M-12) has been fabricated (Fig 120). This vehicle is being used to determine areas of interference between subsystems, to establish the routing of plumbing, waveguides, and wiring harnesses and to develop efficient assembly procedures. The proper thermal control surface finishes have been applied to this mockup.

b. Surveyor wiring harness design. A major consideration in the wiring harness design is the requirement to minimize thermal conductance through the insulation of the thermally controlled compartments. If the copper conductors pass through the compartment wall, the heat loss through the wiring is excessive, so a short section of high thermal resistance wire will be placed in each circuit at the point it passes through the wall. Nichrome V wire has been selected for the purpose.

Because of the dissimilar metals involved, adequate splices between the copper and nichrome wire have been difficult to achieve. Various types of splices, including welding, soldering, and crimping, have been studied, and a four-indent crimp splice, using a short section of crimp tubing, has been selected. The type of crimp tubing has not yet been chosen, and tubing material—both copper alloys and nickel alloys—have been ordered for test.

Each nichrome insert will be treated as follows: stranded, insulated copper wire within the compartment will be crimped to a short length (2.5 to 4.5 in. long) of

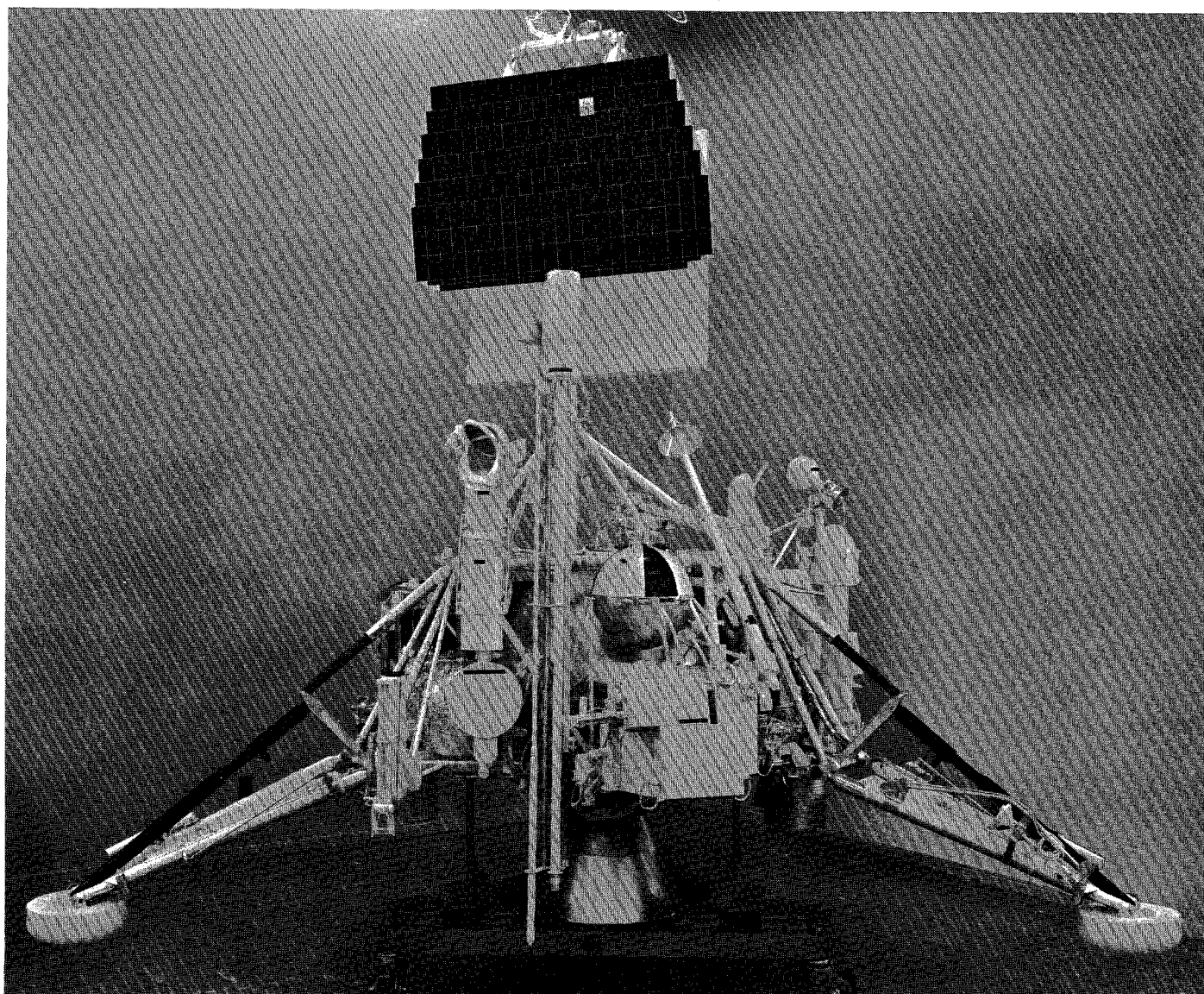


Figure 120. Engineering mockup (M-12) of spacecraft

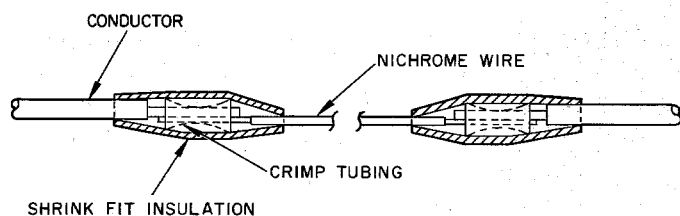


Figure 121. Nichrome insert splicing technique

stranded insulated Nichrome V wire which passes through the compartment wall. An outer crimp joint will be made to splice the Nichrome wire section to the remainder of the harness. The splices themselves will be

insulated with irradiated polyolefin shrink sleeving, overlapping the conductor insulation on either side of the crimp tubing as shown in Figure 121.

Shielded leads passing through the compartment wall must be carefully treated. Where shielding requirements are not critical, a short unshielded length of Nichrome wire will be used. Where shielding is more critical, isolation by interposing Nichrome splices at ground potential will be adequate. For the most critical circuits, a shield braid of Nichrome will be used.

Certain portions of the harness will receive further thermal treatment, consisting of encasing the harness in

a layer of insulation (thermal tunnel) to increase the thermal resistance. This type of thermal control will be used for the wiring that connects Compartments A and B.

During this report period, detail designs of the antenna-solar panel positioner harness magazine, the thermal tunnel interconnecting Compartments A and B, and the altitude marking radar pull-away disconnect have been completed. A lacing cord suitable for the space environment, to be used in lieu of MIL-T-713 types, has been selected, and investigation of a surface treatment for the harness that will be mechanically suitable and meet thermal requirements is continuing.

Fabrication of hardware for the antenna-solar panel positioner harness magazine is in process. As shown in Figure 122, this magazine consists of an open-work aluminum sheet, formed to cover a sector of approximately 200 deg around the mast and supported radially from the mast to leave a 0.5 in. wide truncated annular cross section for cable stowage. Both the positioner drive wiring and the planar array coaxial cable will be stowed in this magazine.

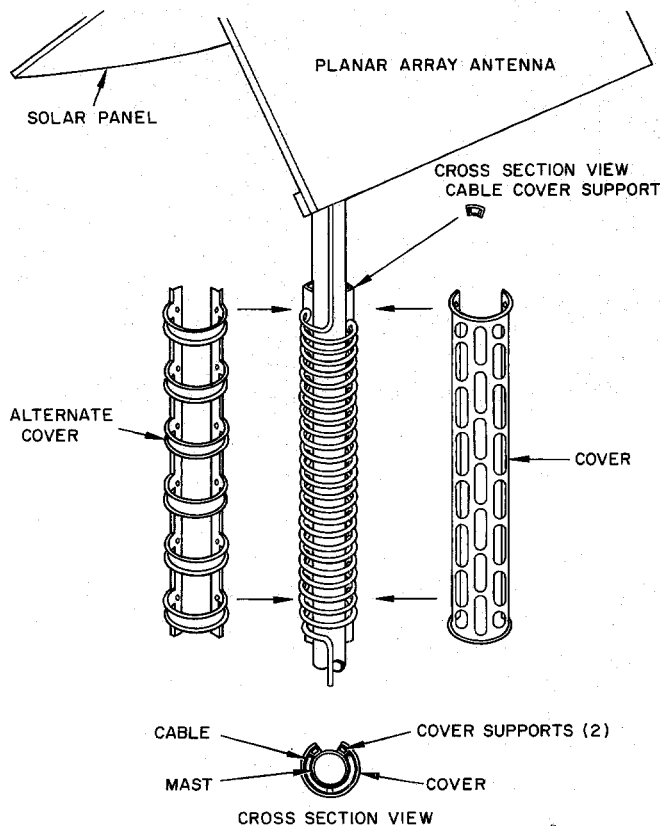


Figure 122. Wiring harness magazine

The thermal tunnel consists of a 3 in. diameter fiberglass tube, supported at three points on the spaceframe and at the entry points of Compartments A and B. The tube serves as a protective cover and support for the harness cabling and its superinsulation wrapping. The tube has a series of convolutions where it crosses the hinge axis of each compartment, to allow the compartment to be rotated out during assembly or disassembly.

A lanyard attachment to the coupling ring of a standard Microdot Series 43 connector has been designed. This connector serves as the interface between the altitude marking radar wiring and the spaceframe harness and must be disconnected upon main retro engine ignition. This connector type requires very low disconnect forces, but tests will be required to assure that tip-off forces do not adversely affect altitude marking radar antenna separation.

Alumina fiberglass tape, manufactured by the Blackwell Division of H. I. Thompson, shows promise of being a suitable harness lacing cord for the space environment. This tape, in $\frac{1}{8}$ in. widths, will be studied for handling ease and resistance to the space environment during fabrication and test of the MT-1 thermal test model harnesses.

c. Landing gear development tests. In the design of the *Surveyor* landing gear, specific elements of the unfolding and locking mechanism depend on test for their final design. These elements are: the landing gear deployment spring, the bearing lubricant, and the downlock damper pad. In addition, tests are required to ensure the adequacy of the downlock locking spring, pin puller bracketery, and downlock signal switch designs.

The tests will be performed on L-1 and L-2 which will consist of actual flight-type landing gear assemblies. The L-1 and L-2 hardware and their test fixtures will be completed by March 2, 1962. The tests will be conducted within the subsequent 6-wk period.

d. Special test vehicles. Fabrication of T-1, the landing dynamics test vehicle, shown in Figure 123, has been completed, and the vehicle is now being instrumented. The side frame mounting provides for alteration of mass properties during the testing period.

Spaceframes and attach structure elements are in fabrication and assembly for MT-1 Sector (3) (thermal control test), S-2 (structure test), and S-6 and S-7 (vernier engine qualification test vehicles).

Preliminary design of T-2, the flight control system test vehicle, has been completed and detail design is in progress.

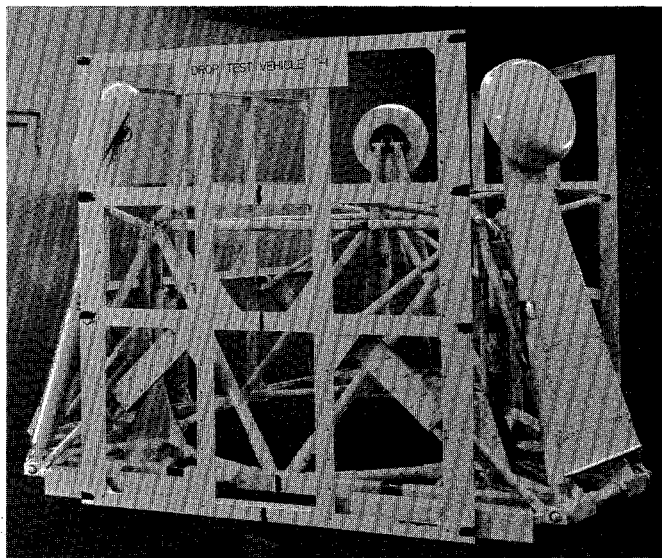


Figure 123. T-1 landing dynamics test vehicle

e. Centaur mockup. On January 15, 1962 General Dynamics Astronautics delivered, on schedule, to Hughes Aircraft Company the following *Centaur* mockup items: (1) nose fairing, (2) forward bulkhead, (3) spacecraft adapter, and (4) tooling gage. The first three items are shown in Figure 124.

Associated drawings for these items were also delivered. The tooling gage was used on arrival to check the adapter hole pattern of the MA-1 antenna model, M-12 full-scale engineering model, and T-1 drop test model. The tooling gage will also be used to make a drill fixture for the *Surveyor* adapter and rocket engine installation.

On February 6, 1962, the MA-1 antenna test model was mated to the forward bulkhead and spacecraft adapter. No major problems were encountered during the initial mating.

2. Basic Bus Mechanisms

Antenna positioning mechanisms and others associated with the basic spacecraft bus are discussed in this section. Mechanisms used in the scientific payload are covered in Section B.

a. Omnidirectional antenna mechanism. This mechanism positions the omnidirectional antenna in the required position after separation of the *Centaur* shroud. The decision has been made to incorporate in the spacecraft the non-telescoping antenna mechanism design reported in the previous bimonthly progress summary and a design layout is in progress. This extension mechanism

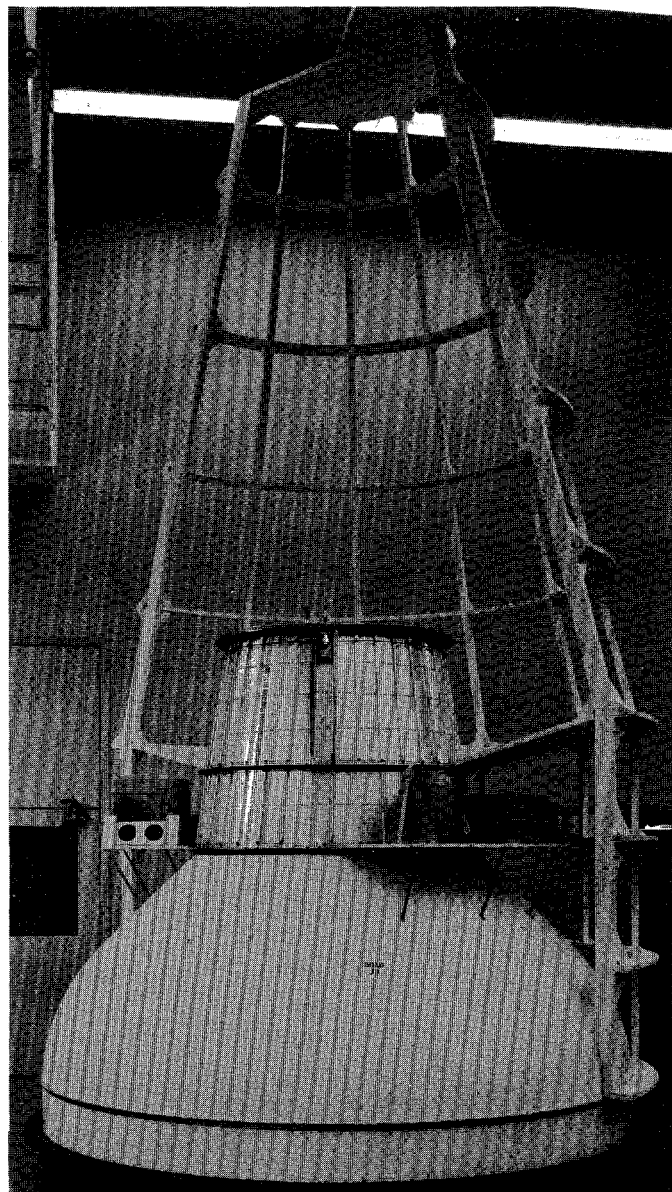


Figure 124. Mockup of *Centaur* forward bulkhead, spacecraft-adapter, and nose fairing

will consist of two similar, but not identical, tapered aluminum tube sections approximately 5-ft. long, one for each antenna and associated release and locking mechanisms. The design of Omnidirectional Antenna No. 1 is shown in the extended position in Figure 125. The extension mechanism and antenna assembly will be stowed in a nearly vertical position on the spaceframe and the mechanism is released by the use of a squib-operated release lock. The antenna and tube are then spring-driven to the operating position where the assembly will remain locked throughout the transit and landing

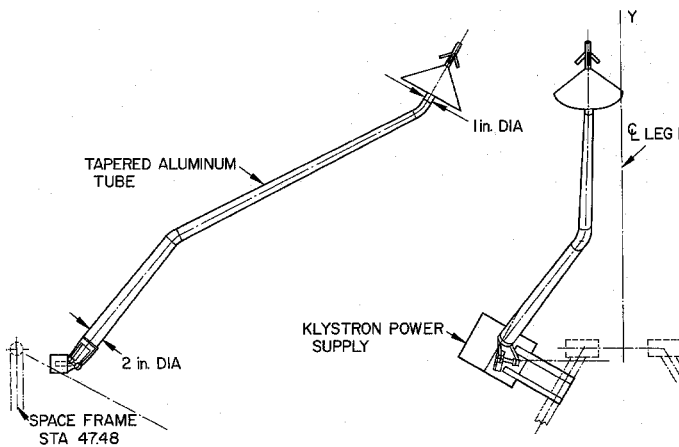


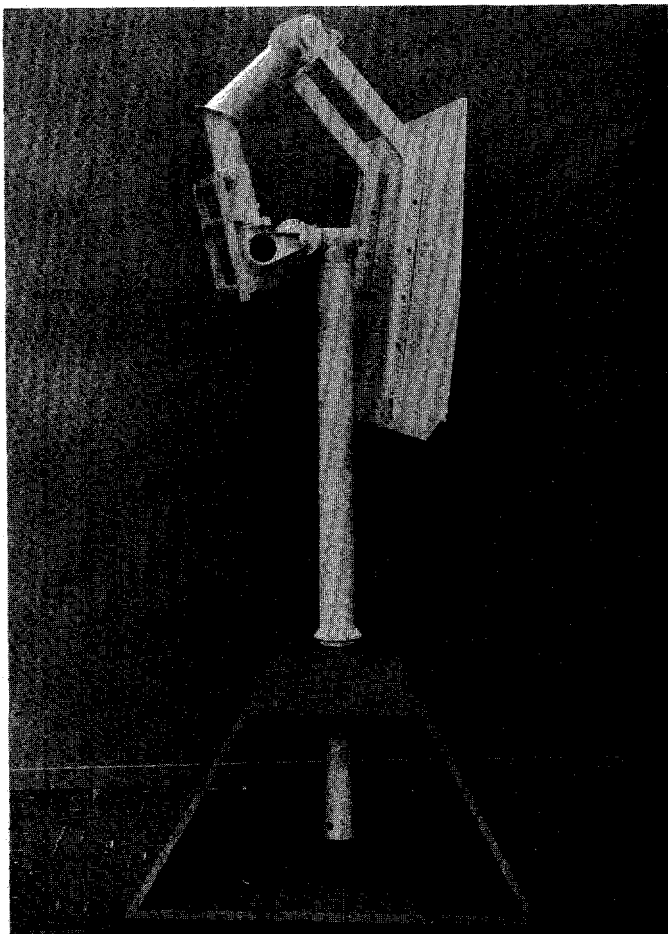
Figure 125. Omnidirectional antenna extended

periods. The orientation of the two mechanisms is such that an unobstructed signal pattern is assured regardless of spacecraft orientation. This design also minimizes the likelihood of a coaxial cable deployment problem, be-

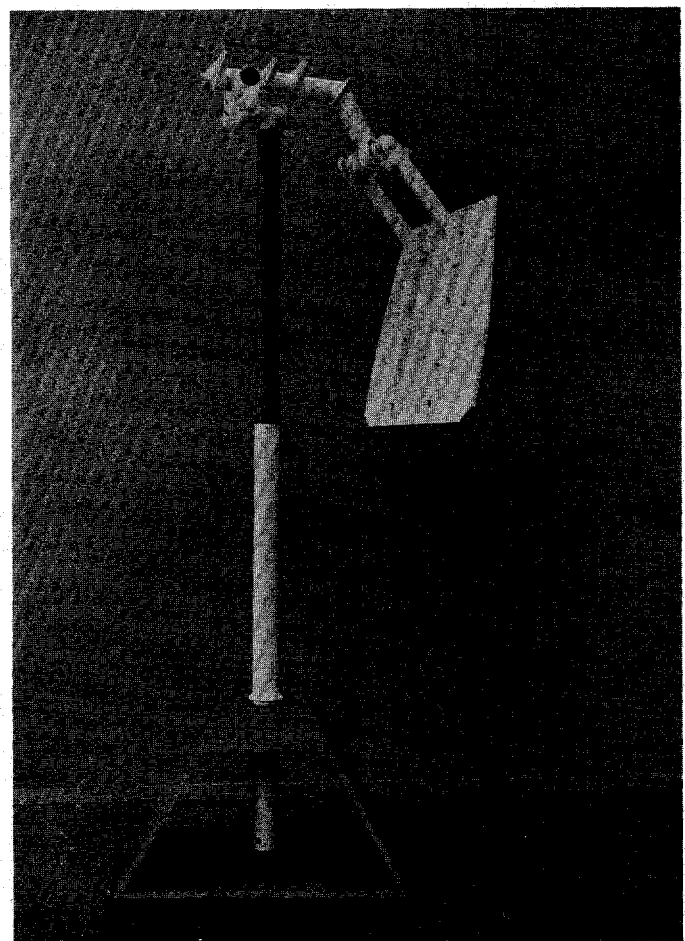
cause the cable will be contained within the tubes and only a single bending movement is required of the cable.

b. Antenna-solar panel positioner. Drawings of the redesigned antenna-solar panel positioner with four controllable axes, described in the previous bimonthly progress summary, have been released for fabrication of an engineering test unit. The earlier-design five-axis developmental model has been fabricated and developmental model tests have begun. The engineering mockup of the five-axis design, shown in the stowed and operating positions in Figure 126, is being used to resolve wiring harness and coaxial cable manipulation problems.

A test fixture for the polar axis worm drive has been fabricated (Fig 127). An off-the-shelf worm and gear assembly is being tested to determine the magnitude of shock and vibration environment effects. Those tests will provide useful information prior to delivery of the worm and wheel assembly fabricated to the latest design drawings.



a) Stowed



b) Extended

Figure 126. Mockup of antenna-solar panel positioner

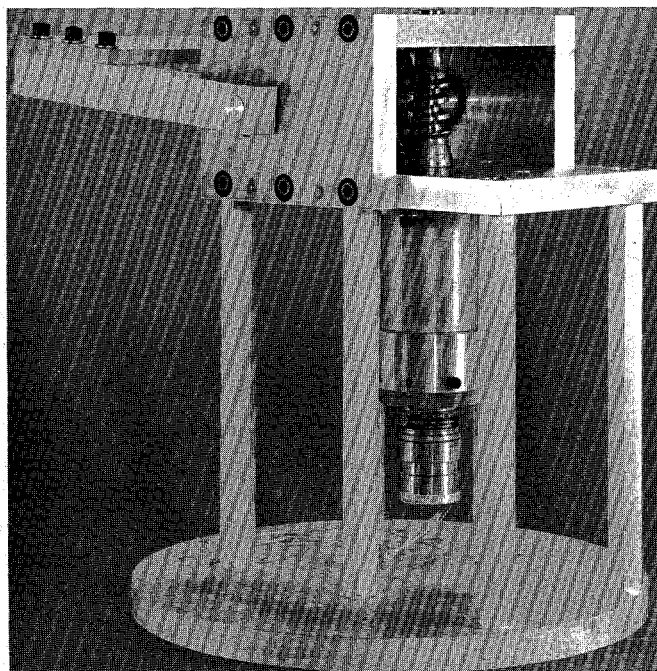


Figure 127. Worm drive test fixture



Figure 128. Vacuum testing of position potentiometers

Resolution and linearity testing of axis position potentiometers is being conducted in a vacuum environment (Fig 128). At pressures of 3×10^{-8} mm Hg and at ambient temperature, no degradation of potentiometer performance has been observed. High-temperature vacuum tests are scheduled to follow the present tests.

c. Motors and mechanisms drive units

Rotary solenoids. A series of high temperature-vacuum tests have been concluded on three modified rotary solenoid type motors (Fig 129). The tests were conducted in a vacuum environment of 10^{-6} to 10^{-7} mm Hg at sustained temperatures of 270° F. Calibration curves of average torque as related to response time were run at ambient temperature and pressure (Fig 130). Response time measurements were then made in the temperature-vacuum environment (Fig 131).

High-temperature failure occurred in two of the motors after 5 min continued pulsing at the rate of 2 pulses per sec. Both motors contained an O-ring bearing which is not adequate for this environment. In addition,

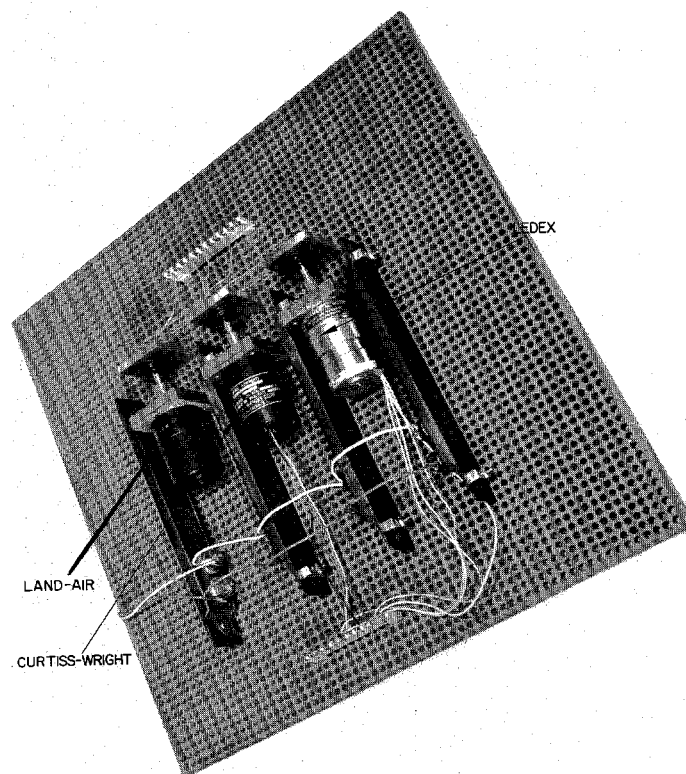
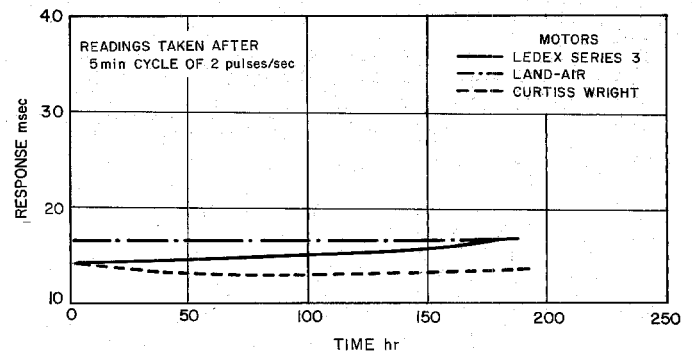
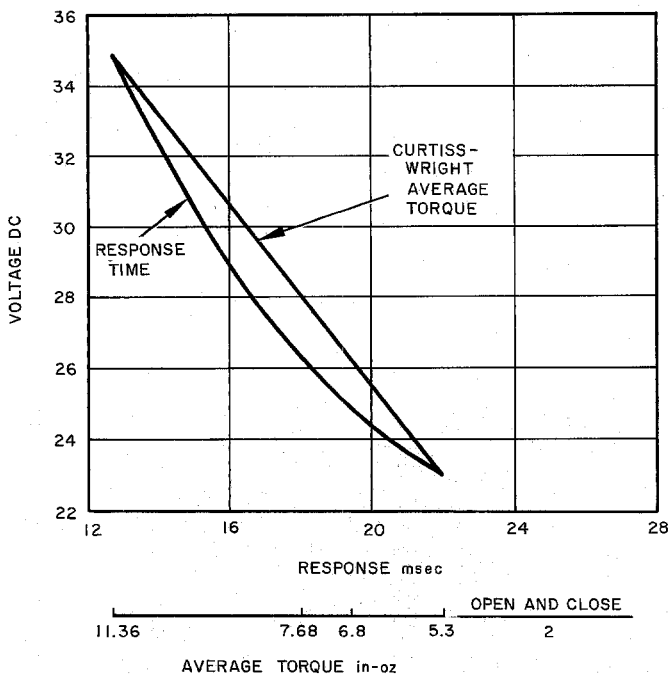
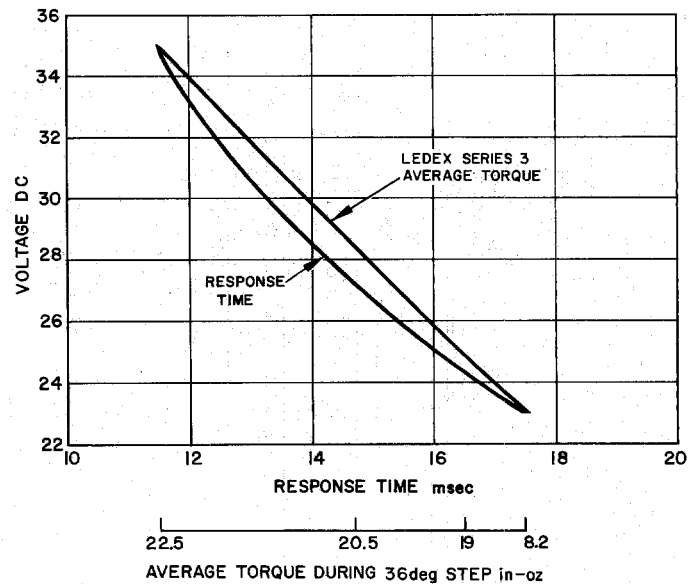
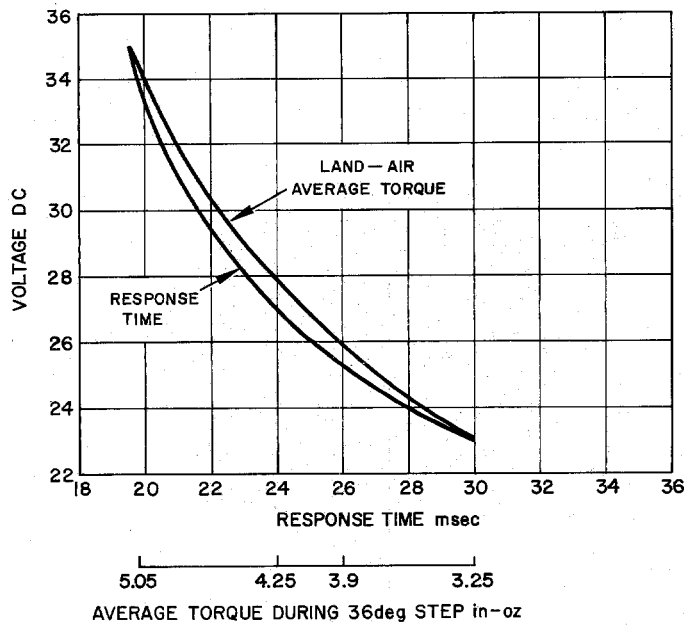
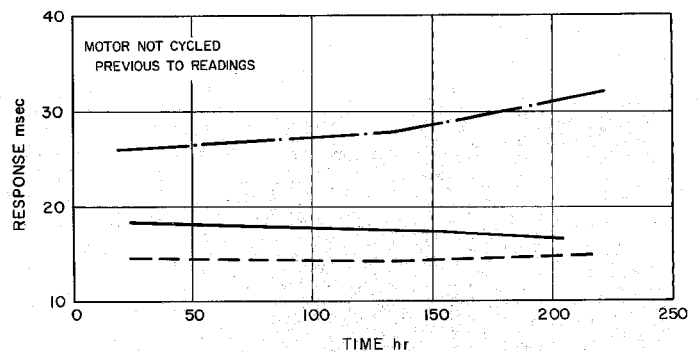


Figure 129. Fixture for high-temperature vacuum tests of rotary solenoid motors



a) Temperature: 45 to 80°F
Vacuum: 2.5×10^{-6}



b) Temperature: 245 to 275°F
Vacuum: 5×10^{-7}

Figure 130. Rotary solenoid calibration curves

Figure 131. Rotary solenoid response time in temperature-vacuum environment

one of the failed motors contained a gold-plated ball and ramp assembly. The gold plate flaked and wore under load, contributing to the failure. The failed com-

ponents are now being analyzed. The tests indicated that bearings of "Flourosint" or steel ball bearings with teflon separator and MoS_2 lubricant are satisfactory for operation in temperature and vacuum environments encountered.

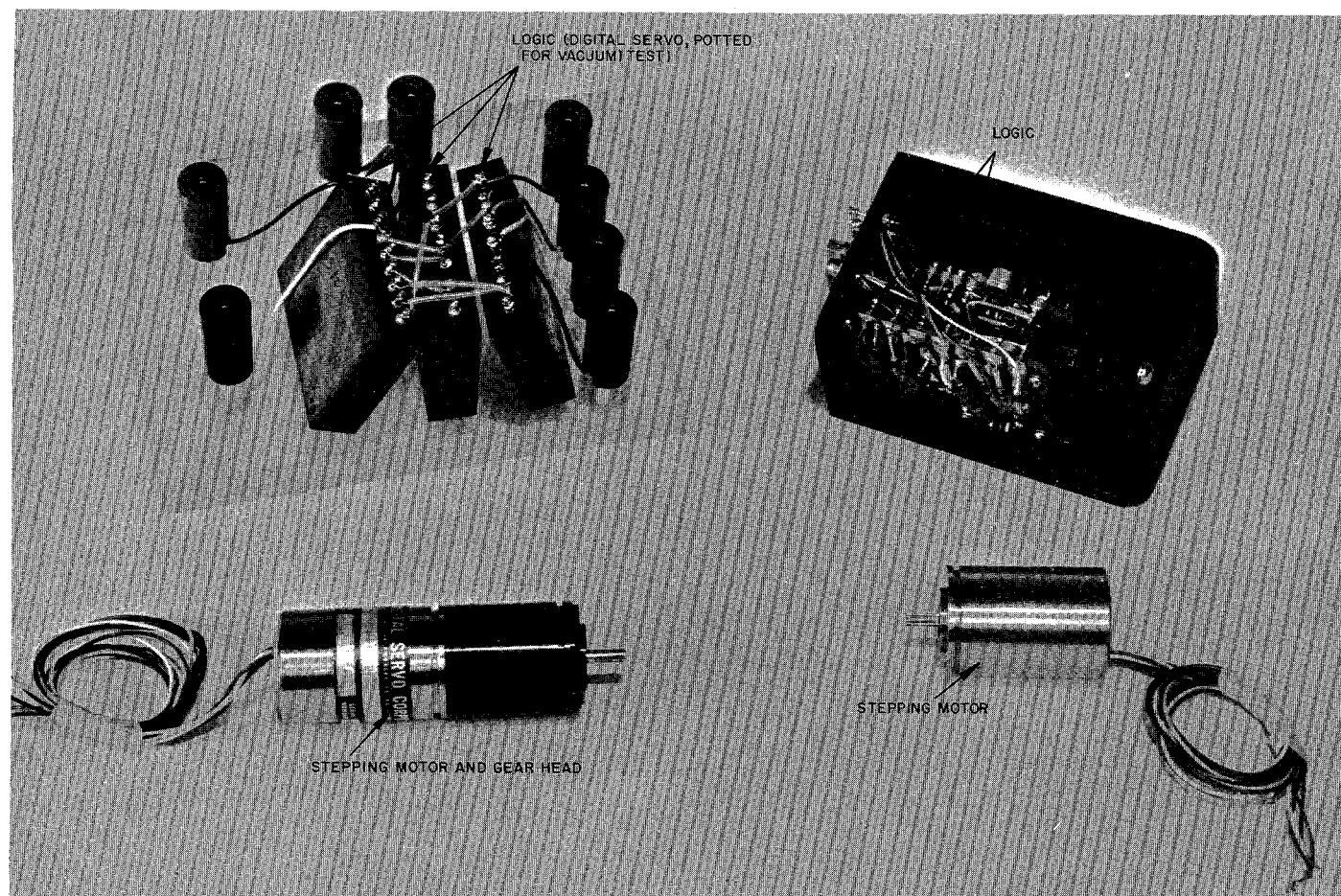


Figure 132. Digital logic phase motors

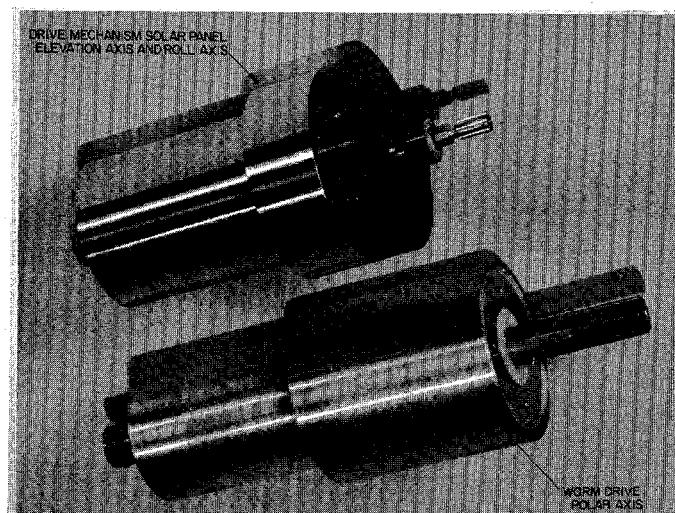


Figure 133. Planetary drive mechanism

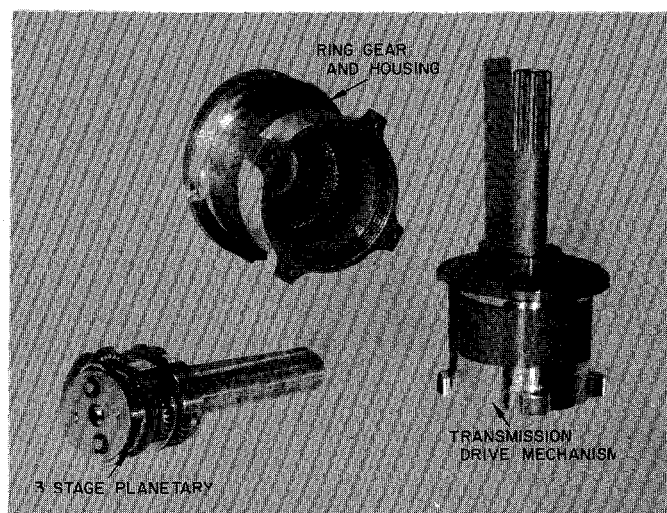


Figure 134. Planetary drive mechanism

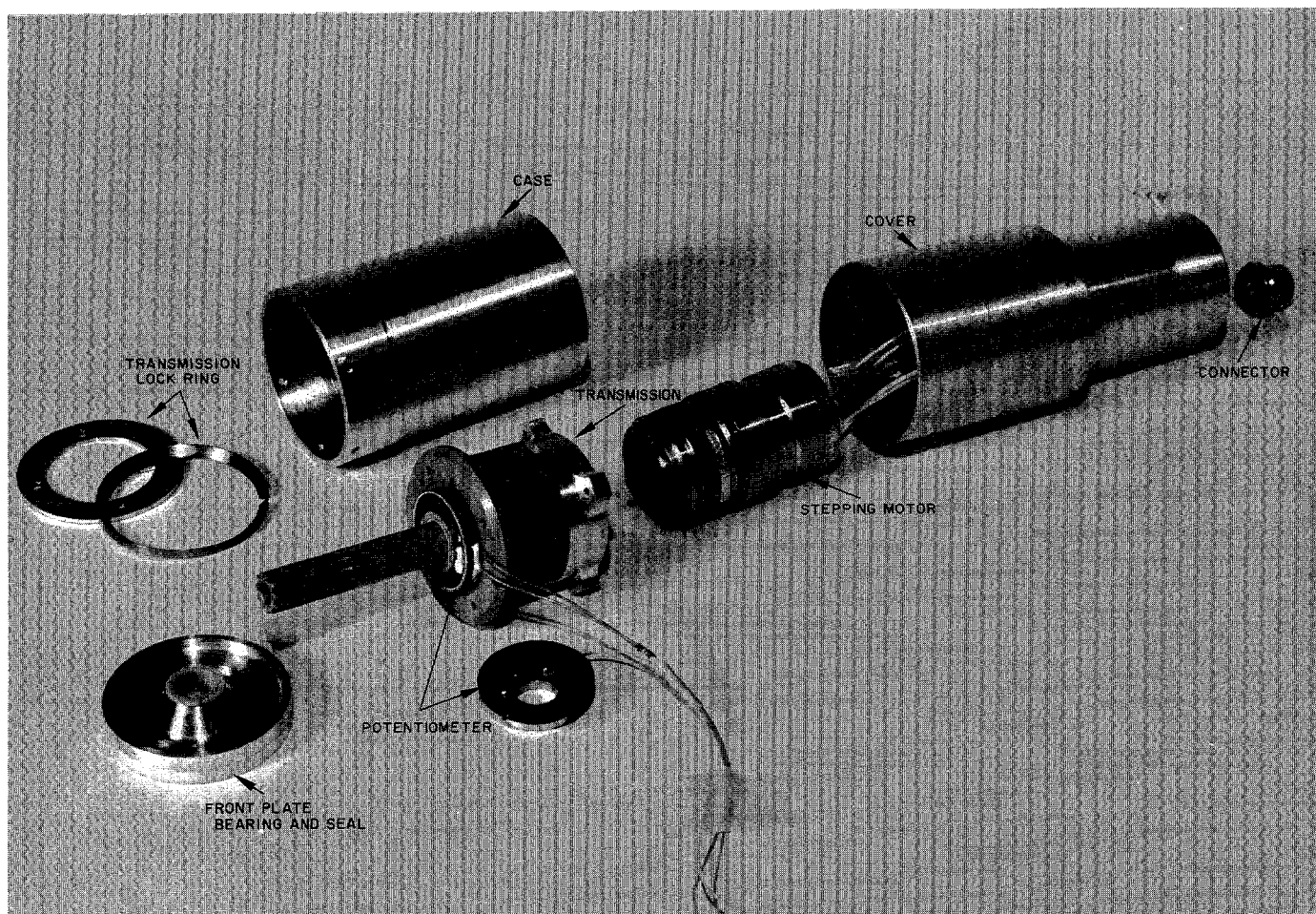


Figure 135. 288:1 drive mechanism, solar panel, elevation and roll axes

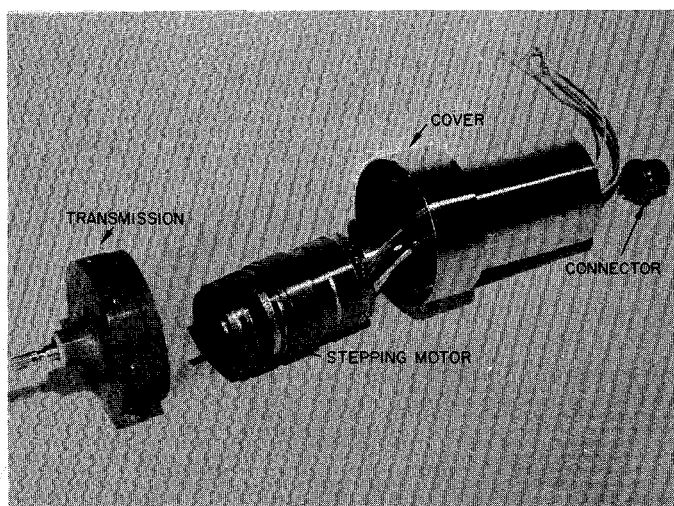


Figure 136. 12:1 worm drive, polar axis

Program pulse phased motors. Two sets of program pulse phased motors and associated logic have been received and evaluated for performance under ambient conditions (Fig 132). This type of motor is required for rapid traverse action in such applications as the high-resolution telescope focus drive.

Drive Mechanisms. A three-stage planetary drive mechanism has been designed and fabricated for use in the roll, elevation, and solar panel axes of the antenna-solar panel positioner (Fig 133, 134, and 135). A single-stage reduction drive mechanism for the polar axis worm drive of the antenna-solar panel positioner has also been designed and fabricated (Fig 133 and 136). These mechanisms are now ready for environmental and load testing in conjunction with their associated drive motors.

d. Separation sensing and arming device. The separation sensing and arming device consists of a double-pole,

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double-throw mechanical switch actuated by the separation motion of the interconnecting structure of *Centaur* launch vehicle from the *Surveyor* spacecraft. Actuation of the switch closes the circuits to the spacecraft explosive devices. For reliability there are two arming devices connected in parallel. A third identical unit provides telemetry indication of separation.

In conjunction with the separation sensing and arming device, an overriding switching circuit permits pre-separation operation of the omni-directional antenna

mechanism and spacecraft landing gear. Prior to assembly to the *Centaur* interconnect structure, the separation sensing and arming device maintains the explosive circuits in the open condition by means of a locking clip which restrains the spring-loaded switch-actuating plunger. Removal of the locking clip after *Centaur* spacecraft assembly permits actuation of the switch circuits at separation. Preliminary testing of switch operation in a vacuum environment has been successfully completed, and fabrication of a complete developmental device is under way.

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K. Reliability and System Test

1. Reliability

a. Spacecraft reliability estimate for flight and landing phase. The following is the third estimate of the reliabilities of the spacecraft system and basic subsystems hardware. The spacecraft reliability for the flight and landing phase is defined as the probability that the spacecraft, after successful injection, will not malfunction to preclude soft landing successfully on the prescribed portion of the Moon.

The functional requirements of each subsystem were synthesized into a system mathematical model which provided the basis for this prediction. It is currently estimated that there is an 80% probability that the system reliability will be greater than 0.57 during the flight and landing phase of the spacecraft. It is further estimated that there is a 20% probability that a reliability greater than 0.86 will be achieved. The mean reliability for the system is estimated to be 0.75.

A summary of the reliability estimate is shown in Table 26. The propulsion subsystem estimate reflects an improvement over the previous report due to the additional information obtained from the detailed analysis of individual engine system reliability received from the engine vendor. The vehicle group has shown a marked decline in reliability due to analysis of a more complete and realistic wiring harness and interconnections schedule.

Table 26. Reliability estimates for flight and landing phase

Equipment	R ₉₀ , %	R ₅₀ , %	R ₂₀ , %
1.0 System — Total	57.3	75.0	85.6
2.0 Flight Control Group Attitude reference sensors, control equipment, roll vernier actuator, retro burnout sensor, inertial reference unit, attitude jet system, solder joints.	74.6	90.3	96.0
3.0 Propulsion Group Vernier engine system, retro engine system.	85.4	93.9	97.7
4.0 Electrical Power Group Solar panels, batteries, solder joints.	83.0	95.6	98.9
5.0 Electronics Group Power system control, command receivers, transponders, telemetering transmitters, central command decoders, signal processing circuitry, radars, TV 4 electronics, solder joints.	90.6	95.0	97.3
6.0 Vehicle Group Harness and interconnections, retro-rocket separation, vehicle structure, landing gear.	77.5	96.2	99.4
7.0 Mechanics Group Antenna and solar panel drive, vehicle anchoring, TV 4 manipulating devices and optics.	99.4	99.7	99.9
8.0 Thermal Control Group Sensors	99.3	99.9	>99.9

This decrease is due mainly to the firm requirement for a large number of connector pins and solder joints not originally accounted for. Similarly, inclusion of the latest design information has increased the number of solder joints planned in the electronics group and has caused the indicated reliability estimate to be slightly lowered. The remaining equipment group reliabilities have remained essentially unchanged during this reporting period.

The system reliability equations for the lunar operation are being developed and are anticipated to be completed during the next reporting period.

b. Quality assurance. The quality assurance program has been prepared and submitted for JPL approval. Additional definitions in the areas of failure review board, first-article inspection, and materiel handling and storage project bulletins are items requested which are in the process of being defined during the next reporting period and will be incorporated as they become project-approved.

c. Components and materials. Under study at this time is a plan to include conditional approval for certain components and materials which cannot survive the full lunar environment but may survive the particular environment of the particular spacecraft installation within which they are located. It is anticipated that a conditional qualified family of components and materials will be made available through this program which will eliminate attempting to qualify many alternate types or require extensive circuit redesign to utilize only fully qualified components. It is expected that spacecraft sector requirements and an indication of number and types of components that may be allowed conditional or limited environment qualification will be available for the next Bimonthly Progress Summary.

2. System Test

a. Equipment design. The functional requirements for the command decoding and signal processing elements of the system test equipment assembly (STEA) have been developed (Fig 137). The command decoding, scientific signal processing, and engineering signal processing functions of the spacecraft will be qualification-tested. Equipment for this purpose will be the command encoder, discriminators, decommutator, digital-to-analog converters, and tape recorder of the command and data handling console (CDC) and two test racks of additional equipment.

The first major form of testing will be group testing.

During these tests, the RF link will be disconnected as will the scientific instruments and engineering sensors. Commands to the spacecraft will be inserted from the CDC by its keyboard, backup switches, or tape reader. The binary coded output of the CDC encoder will be conducted via hardline to the input of the central decoder unit, thus bypassing the RF link. Decoding and selection of subsystems will be accomplished in a normal manner.

Inputs to scientific signal processing and engineering signal processing will be simulated at the STEA and conducted to the spacecraft via hardline. Signal processing through the auxiliaries and central signal processing will be accomplished in the normal manner.

The output of central signal processing will be conducted via hardline to discriminator inputs at the CDC, thus bypassing the RF link. The remainder of the CDC will be employed in the normal manner.

Command decoding. The command decoding function receives commands from DSIF sites and processes this information into a form that initiates performance by selected subsystems within the spacecraft. This information exists in various configurations of pulses; therefore, command decoding will be qualified by examining all significant wave shapes out of the central decoder unit. In addition, every subsystem address will be selected and commanded to apply power. Power application will be observed at a monitor panel.

Commands are applied to the spacecraft via hardline, bypassing the CDC transmitter and the spacecraft receiver. Also, central decoder unit outputs are routed to the STEA via hardline, bypassing the telemetry.

Scientific signal processing. This function of the spacecraft is to process ac, dc, and pulse information from the output of scientific experiments into a frequency modulated form for input to the spacecraft transmitter. Several factors relating to the quality of this processing will be evaluated.

- (1) Linearity testing will verify the operation of voltage-controlled oscillators. It will be established that frequency variation responds to within 0.5% of input voltage variation throughout the voltage range.
- (2) Scale factor testing will verify the operation of amplifiers that are required after the outputs of some scientific experiments. Accuracy requirements will vary from 1 to 15% depending on the experiment under test.

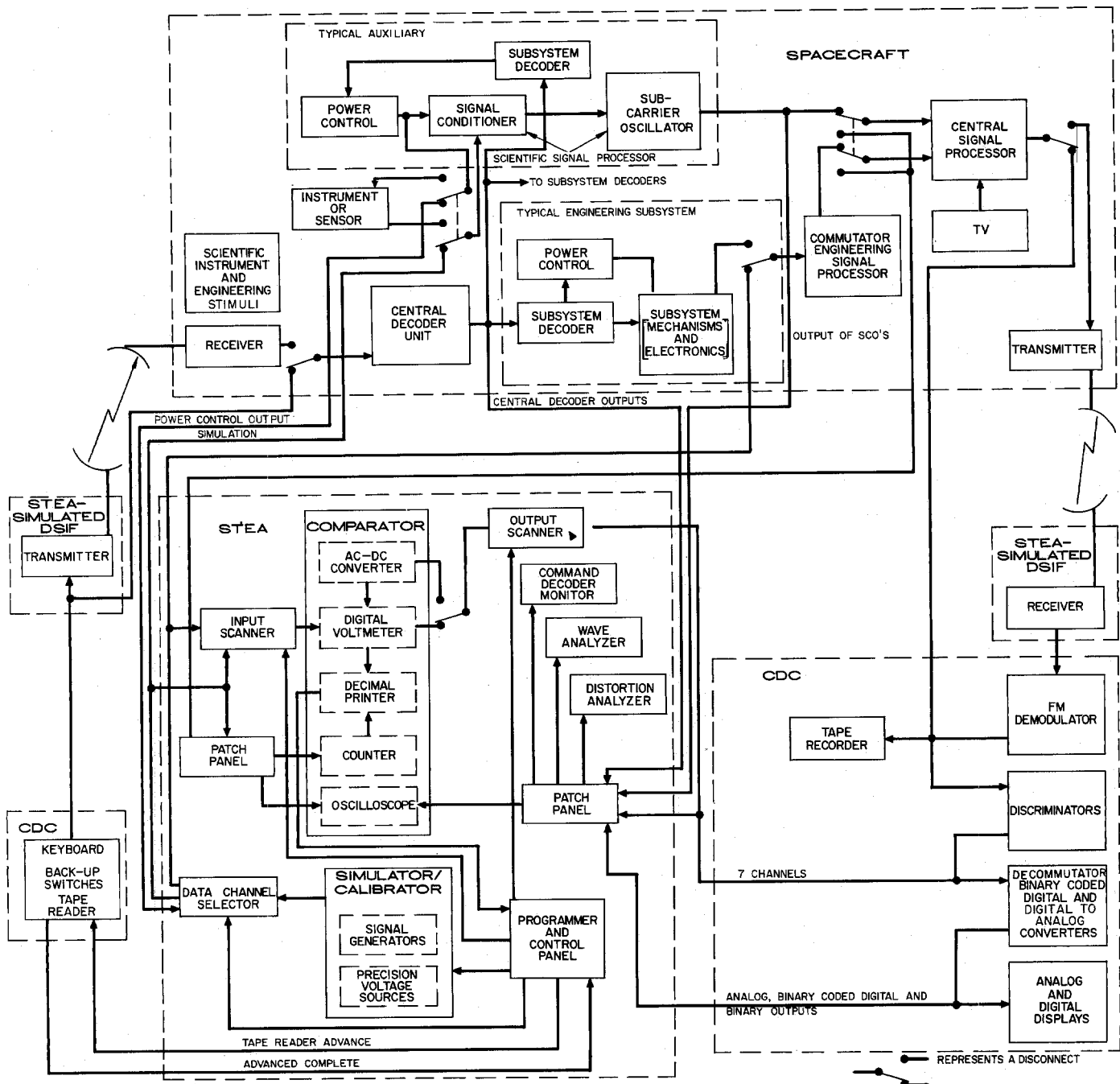


Figure 137. Command decoding and signal processing checkout

- (3) Stability testing will involve application of an input to a data channel that will cause its voltage-controlled oscillator to oscillate at its center frequency. This test will be repeated periodically to establish a frequency versus time relationship and must yield a stability of 0.5%.
- (4) Distortion testing will be conducted in each data

channel. Harmonic distortion will be measured and must be less than 2%.

- (5) Frequency response tests are a requirement for some data channels. Frequency response will be verified by manually varying the input frequency to these channels and measuring the discriminated output voltages. The 3 db points will be deter-

mined. The specified bandwidth for each channel test will be verified.

- (6) Intermodulation products will be measured by applying a signal to a data channel and discriminating both the fundamental voltage-controlled oscillator frequency and its third harmonic. The two values will be compared to determine that the third harmonic is down 40 db from the fundamental.

Power will be applied to data channels by command from the CDC through the central decoder unit in the spacecraft. Input simulation will be applied via hardline to the inputs of scientific experiment auxiliaries (outputs of scientific instruments). Outputs will be taken from the output of signal processing (input to the spacecraft transmitter), hardlined to the CDC to be discriminated, and applied to the STEA for measurement.

The spacecraft tape recorder records information in one direction and plays back in the opposite direction. It will be qualified by operation in its normal manner. The telemetered output will be recorded on the CDC tape recorder. The data processing facility will then be employed to evaluate the performance of the equipment.

Engineering signal processor. This spacecraft function is concerned with all basic bus quantities and commutated scientific experiment data. DC analog and on-off data are processed through commutators and voltage-controlled oscillators to the spacecraft transmitter. Other dc analog and pulse data are applied directly to other voltage-controlled oscillators. The significant parameters of this function will be qualification-tested.

- (1) Commutation: DC voltage inputs will be applied to the engineering signal processor such that the grating circuits that comprise the commutator will be exercised. The commutated data will be decommutated at the CDC and compared with the original inputs. This test will be similar to the linearity testing of scientific signal processing.
- (2) Amplification: Some engineering quantities are amplified within the spacecraft. Small dc voltages will be supplied by the STEA and applied to the amplifier in the spacecraft. The amplified outputs will be routed through the CDC to the STEA where the amplification factor will be qualified.
- (3) Analog-to-digital conversion: Different dc voltages will be applied to the engineering signal processor. The decommutated outputs will be observed in their binary configuration. The binary configuration must compare with the input analog configuration to within 0.1%.

- (4) Commutator timing: The frequency of the clock in the analog-to-digital converter will be qualified by applying zero volt inputs to the spacecraft and by counting the parity pulses that return. This count must compare within 1% of the specified clock frequency for the four modes of commutator operation.

Control over engineering signal processing is exercised via the command decoding function. Simulated inputs are applied at the input to the engineering signal processor while outputs are taken at the output of signal processing to the CDC for discrimination, decommutation (if necessary), and application to the STEA.

Conclusion of group testing will be followed by system testing, which will sequentially remove hardline connections and incorporate spacecraft power, the RF link, actual engineering data inputs, and actual scientific experiment inputs. The tests that are performed during group testing will then be repeated, with the exception that inputs to data channels will be generated by stimulation of their sensors instead of by simulation from the STEA.

b. Magnetic test facility. The search for a magnetic test facility site has narrowed down to two locations. One potential site is in Baldwin Hills and the other, in Topanga Canyon. The Baldwin Hills site has been surveyed for dc magnetic field gradient and was satisfactory, being less than 1 gamma per foot. A 3-ft-diameter search coil to provide increased sensitivity is being constructed for use in measuring the alternating magnetic field. The coil output can be connected directly to the input of a sensitive battery-operated VTVM, making the use of an additional special preamplifier unnecessary. Immediately upon completion of the search coil and delivery of a battery-operated VTVM, measurements of the alternating magnetic field at Baldwin Hills will be made. Pending this survey preliminary negotiations effecting a lease on this property are progressing. The proximity of this site to Hughes Culver City makes it much more desirable than the Topanga Canyon site.

The detail specification on the nonmagnetic building has been completed and released. A purchase request for the magnetic coil array, along with specifications and drawings, has been presented to JPL for approval.

c. Environmental testing. During this period component testing was initiated in the vacuum chambers of the Environmental Test Facility. Hi-Vac I was utilized for testing of thermal switches and in three phases of compartment radiation insulation testing. A series of

tests of stepper type motors which will be used in various spacecraft mechanisms were performed in Hi-Vac II.

Two additional chambers were recently delivered and are now undergoing acceptance tests. Hi-Vac III, a bell-jar with 1×10^{-9} mm Hg capability, is operational. Hi-Vac IV is expected to be operational February 15 under ambient temperature conditions to a pressure of 1×10^{-7} mm Hg. The liquid nitrogen shroud is to be reworked and is expected to be available about March 15.

Construction of the Space Simulation Laboratory is continuing; the mezzanine structure for Hi-Vac V and VI is shown in Figure 138.

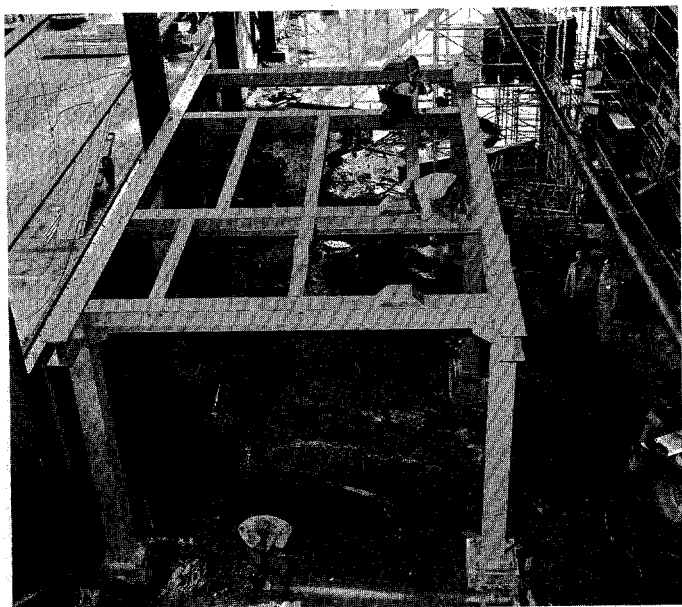


Figure 138. Construction of Space Simulation Laboratory

d. Control system vehicle, T-2. The testing of the spacecraft flight control system terminal landing phase will be accomplished on a modified vehicle frame designated T-2, at the Air Force Missile Development Center (AFMDC), Alamogordo, New Mexico. The present scheduling calls for this to be conducted at AFMDC from October 1 to December 15, 1962.

General objectives. The following general test objectives have been established for the T-2 flight control vehicle:

- (1) Perform an over-all dynamic test of the final portion of the vernier descent phase, including actual touchdown, in as near lunar conditions as possible.

- (2) Determine the capability of the vernier engine system and the attitude reference system to hold vehicle attitude during dynamic descent under varying vernier thrust conditions.
- (3) Determine the capability of the vernier engine system and the doppler velocity sensor to correctly align the thrust velocity vector under dynamic conditions with various initial velocities along each axis.
- (4) Determine the capability of the system to attain touchdown conditions using representative velocity attitude and attitude control equipment.
- (5) Gather reliability data on spacecraft components on a ride along basis.

Vehicle configuration. The vehicle configuration consists of a modified pyramid frame (Fig 139) upon which is mounted the vernier engine system, flight control group, power source, and recovery system. The recovery system will consist of a parachute-airbag combination. The parachute will be initiated at engine shutdown by the program sequencer. A command link override is provided to allow recovery initiation at any time during the tests. The parachute, after recovery initiation, will stabilize the descent to approximately 18 ft/sec. The remaining energy will be absorbed by an air-bag landing gear. The angular acceleration capability of the T-2

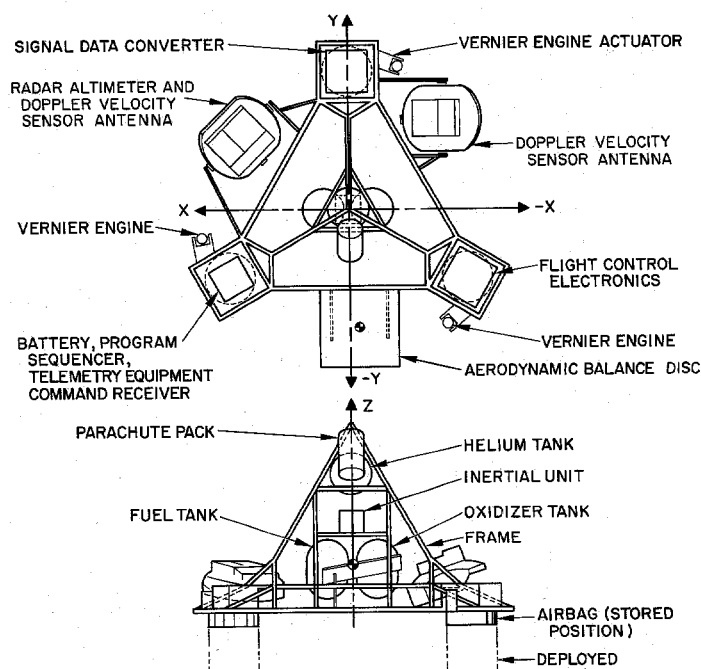


Figure 139. T-2 test vehicle

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vehicle will not be scaled to that of the lunar design. Artificial stabilization in the form of a small drag chute will be employed in all tests in the velocity region above 20 ft/sec. Prototype radar and electronics units will be utilized.

Test procedures. The philosophy of the test sequence is to test progressively larger portions of the system until the complete system has been tested. Reliability data will be recorded during the entire test program. This test program has been divided in four groups.

Group I - Tethered hover tests. This group of tests is comprised of two separate tests during which the vehicle will be suspended from a balloon for the entire test periods.

Test 1: Utilizing the inertial hold loops for attitude control, the vernier engines will be started and thrust will be varied from minimum to maximum. The objectives of this test are: (1) general engine operation, ignition, thrust variation, and shutdown; (2) test the stability of the inertial hold mode under varied thrust conditions, check individual engine gain variation and alignment effects on inertial hold mode; (3) check for effects of engine-induced vibration by monitoring all telemetered signals; and (4) Test sequence timer and emergency recovery operation.

Test 2: With the system in inertial hold mode and the engine thrust set at minimum and maximum levels, rate commands will be applied individually and in combinations. The objectives of this test are: (1) to compare actual response with theoretical response, and check the coupling of the vibrations from the engines through the spaceframe into the inertial gyros; and (2) to gather information for optimizing of modes, if required. System modifications will be made as required before starting the dynamic tests.

Group II - Drop test series, radar attitude control. This group of tests will consist of a series of free fall drop tests in which the vehicle will be released from a balloon and recovery made at the required altitude after completion of the tests.

Test 3 - Radar functional check: With inertial loops closed, drop the vehicle from 1200 ft altitude with an initial vertical attitude and constant thrust of 60 lb. Initiate recovery after 400 ft (8 sec) from balloon release. The objectives of this test are as follows: (1) monitor and evaluate the radar altimeter and doppler velocity outputs over the expected testing altitudes and velocities; (2) measure aerodynamic forces and

moments over the expected velocity range as a reference in data reduction and for possible final trimming of the vehicle; and (3) complete checkout of the emergency and operational trajectory termination procedures.

Test 4 - Radar attitude control: With inertial and radar attitude loops closed, the vehicle will be dropped with an initial vertical attitude and a minimum thrust. After 3.5 sec from balloon release, maximum programmed thrust will be commanded for another 10 sec at which time the engines will be shut down and recovery initiated. The objective of this test is to evaluate the radar attitude control under varied conditions of velocity and engine thrust.

Test 5 - Radar attitude control (nonvertical drop): The test conditions will be the same as in Test 4 except that the vehicle will be dropped from an initial attitude of 10 deg from the vertical. The objective of this test is to evaluate the radar attitude control with initial nonvertical attitude.

Group III - Trajectory control. This group of tests will consist of a series of drop tests which have as their objective the control of the vehicle trajectory.

Test 6 - Trajectory control functional check: Utilizing inertial hold and radar attitude control, the vehicle will be dropped from a vertical attitude and a minimum thrust level. Three sec after balloon release, maximum thrust will be commanded for 16 sec at which time the engines will be shut down and recovery initiated. The objective of this test is to evaluate the altitude-velocity function generator and acceleration sensor.

Test 7 - Acceleration loop: Utilizing inertial hold, radar attitude control, and accelerometer-controlled engine thrust, the vehicle will be dropped from an initial vertical attitude and minimum thrust. After 3 sec an acceleration will be commanded for an additional 16 sec. The engines will then be shut down and recovery initiated. The objective of this test is to evaluate the acceleration loop and interaction between attitude and acceleration controls.

Test 8 - Velocity loop: Utilizing inertial hold, radar attitude control, and with the acceleration and velocity loops closed, the vehicle will be dropped from an initial vertical attitude. A constant 20 ft/sec velocity will be commanded. The engines will be shut down after 9.5 sec from release and recovery initiated. The objective of this test is to evaluate the velocity loop and to analyze interaction of attitude, acceleration, and velocity controls.

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Test 9 — Trajectory loop: With all control loops operating the vehicle will be dropped from an initial vertical attitude and minimum thrust. Engine shutdown and recovery will be initiated at 22.5 sec after release. The objective of this test is to evaluate the complete trajectory control during simulation of the final vernier descent trajectory with the exception of actual soft landing.

Test 10 — Trajectory control (nonvertical drop): The test conditions will be the same as in Test 9 except that the vehicle will be dropped from an initial attitude of 10 deg from the vertical. The objective of this test is to evaluate the trajectory control with an initial nonvertical attitude.

Group IV — Soft landing.

Test 11 — Trajectory control with soft landing: With all control loops operating, the vehicle will be dropped in a vertical attitude from the proper altitude to allow an actual landing. The initial thrust will be 60 lb. The objective of this test is to evaluate the total system performance during the final vernier descent to soft landing.

Test 12: Repeat of Test 11.

During the above tests, specific tests may be repeated where additional data is required. The artificial stabilization, required in the higher velocity regions, will be

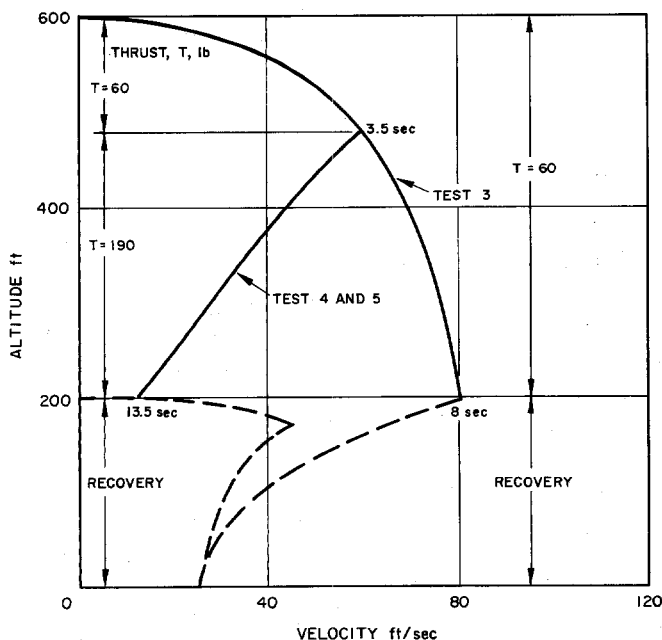


Figure 140. Typical drop test velocity-altitude profile — Group II

jettisoned on each test when the vehicle velocity has been reduced to 20 ft/sec. Typical drop test velocity-altitude profiles are shown in Figures 140, 141, and 142 and grouped according to series.

Suitability of T-2 as a lunar landing test bed. The following analysis illustrates on a comparative basis how each function of the terminal descent flight control operation is evaluated during the T-2 test.

The terminal descent phase of the *Surveyor* mission begins after main retro separation and consists of the following basic functional phases:

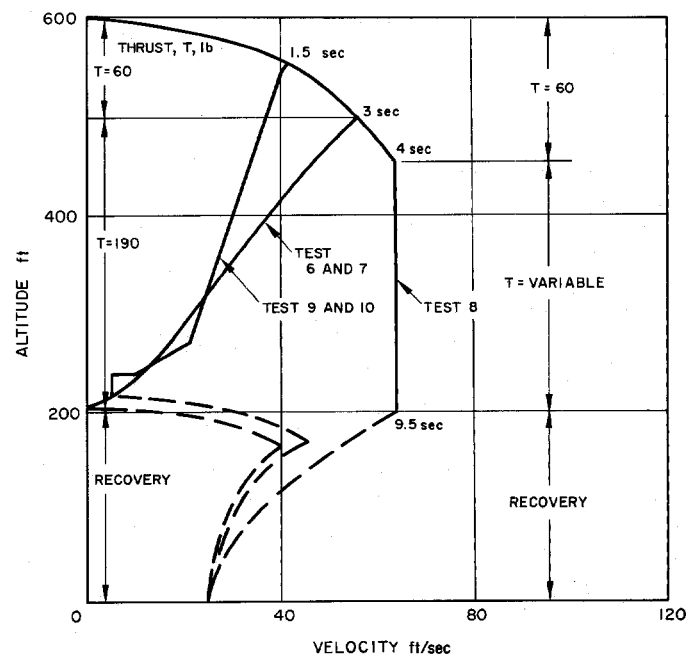


Figure 141. Typical drop test velocity-altitude profile — Group III

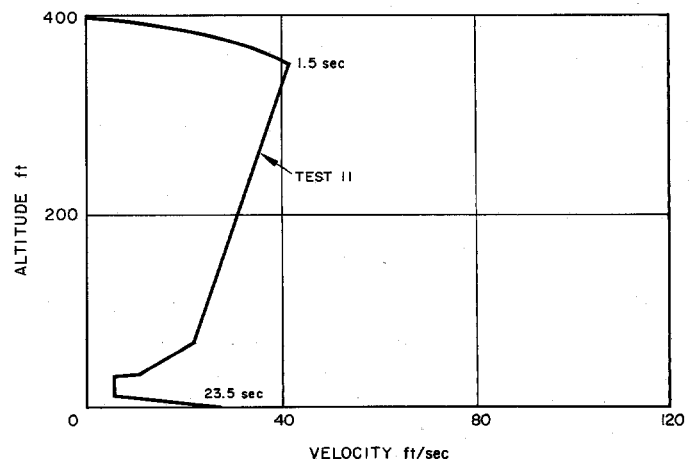


Figure 142. Typical drop test velocity-altitude profile — Group IV

- (1) Vehicle fall in a gravity field with a minimum acceleration command and inertial attitude control.
- (2) Radar operation in search mode until actual doppler lockon is achieved resulting in radar reliable indication.
- (3) Vehicle operation with radar attitude control and the thrust magnitude controlled by the acceleration limiting circuit.
- (4) Intercept of the trajectory curve and the resulting acceleration loop thrust control.
- (5) Operation along the trajectory curve until the 10 ft/sec mark switches the circuit 5 ft/sec constant velocity control.
- (6) Constant velocity drop until 13 ft altitude is indicated from the radar.
- (7) Engine shutdown and free fall to the lunar surface.

During each of these basic functional areas of the terminal descent phase, certain flight control operations are of interest. During the T-2 series of tests all function phases are to be demonstrated.

Each of these phases is discussed and a comparison is made between the actual system performance desired and the degree of evaluation to be accomplished by T-2. First it must be noted that the original plan for T-2 was to keep the ratio of the vehicle of inertia, equal to the flight spacecraft. Due to uncertainties in the aerodynamic roll moments, the roll lever arm was increased to ensure adequate roll moment capability. In order to keep the T-2 configuration functionally equivalent, the electronics gain will be changed to keep all control loop gains equal to those in the spacecraft. A root locus analysis was prepared using the best available aerodynamic data and a comparison of this analysis with a similar one prepared for the lunar vehicle illustrates a very close correlation. By maintaining equal loop gains the dynamic performance of the vehicle can be correlated with a high degree of confidence.

Phase 1 – Vehicle fall in a gravity field with a minimum acceleration command and with inertial attitude control. For the *Surveyor* lunar vehicle the items of interest during this phase are: (1) the inertial attitude control capability in terms of dynamic response to disturbances and the steady-state error of the attitude loop; (2) the acceleration loop stability and summing operation; and (3) vibration feedback into the accelerometer.

Each of these items will be demonstrated with the exception of the steady-state error of the attitude loop. This error will be influenced by the aerodynamic moments introduced in the T-2 test and correlation can be shown only if accurate aerodynamic data are available. The data could be obtained only by extensive wind tunnel tests. Even with the data it is not certain, because of aerodynamic cross coupling, that a steady-state condition will be obtained. The acceleration loop stability will be demonstrated but not at the same low acceleration level planned for the flight spacecraft. The aerodynamic drag forces of T-2 will be measured by the accelerometer in addition to the thrust term. Since drag will increase with velocity, the thrust required will decrease for a given acceleration limit. If the same low acceleration value planned for lunar operation were to be used for T-2, the thrust required would be below linear range of the vernier engines. The result would be a limit cycle attitude control operation. Since this is not desired, the acceleration loop will be tested at a higher value of acceleration.

The vibration feedback to the accelerometer will be evaluated in the T-2 vehicle. Although the vibration transmission paths are different from the flight spacecraft some vibration is anticipated. The noise thus introduced to the accelerometer signal and its affect on this control loop will be evaluated.

Phase 2 – Radar search and reliable indication. During the retro phase of the descent, the velocity sensing radar and the radar altimeter will be searching until the proper range for lockon. It is therefore desirable to be able to operate the radar in a search and lock operation and this is actually accomplished in the T-2 tests. Although the radar will be in range, searching will be required to meet the velocity requirements. The T-2 vehicle starts with an initial zero velocity and thus there is no doppler shift and the radar continues to search. It is only after obtaining a certain minimum velocity that the doppler shift can be detected for lockon. This lockon operation will be evaluated and therefore the T-2 operational check will increase confidence in the radar reliable operation and the resulting switching operation.

Phase 3 – Radar attitude control – During the descent phase as soon as radar reliable occurs, the radar attitude control operates to align the vehicle thrust axis with the vehicle velocity vector. The result of this operation is to reduce any lateral velocity component. This function is being accomplished in T-2 operation by introducing a lateral velocity prior to radar reliable operation and then observing the stability and response of the attitude

loop after radar is reliable. The accuracy and steady error of this loop can be determined to a high degree of accuracy with a correspondingly high confidence in proper performance.

Phase 4 — Intercept of the trajectory curve. The main interest associated with this phase is capability of the system to decelerate and capture the trajectory curve. This evaluates the radar outputs and the radar thrust control loop response. The T-2 test will simulate this action. The deceleration rate of the T-2 is not as great as for the lunar operation because of the reduced thrust of the vernier engines due to atmospheric pressure and it has also not been possible to reduce the T-2 weight to fully compensate for the greater Earth gravity. Although the deceleration rate is lower, the trajectory capture characteristics including overshoot will be similar enough to lunar operation that this phase can be verified accurately.

Phase 5 — Operation along trajectory curve. The critical areas associated with this phase are at the point of transfer from one trajectory line segment to the next. It is necessary to evaluate how quickly and how accurately the new trajectory line segment is followed. All of these operations are being accomplished by the T-2 test. It might be added that due to the aerodynamic disturbances expected for T-2, adequate performance during T-2 test will be a more severe test than for lunar operation, thus greatly improving confidence in lunar capability.

Phase 6 — Constant velocity operation. During the lunar operation the constant velocity of fall is at 5 ft/sec and the transition from trajectory control to constant velocity control is of considerable interest. The T-2 test will verify constant velocity control first at a velocity of around 20 to 30 ft/sec and then at the actual 5 ft/sec operation on subsequent tests. The 20 to 30 ft/sec. test will allow a chance to evaluate the velocity loop for a longer length of time than for the *Surveyor* lunar drop. This higher velocity test will also impose a more severe operating condition due to higher aerodynamic forces. Adequate T-2 operation will ensure proper operation for the less severe case on the lunar surface. The actual 5 ft/sec test will be the concluding part of each trajectory control test and will be used as part of the actual soft Earth landing test.

Phase 7 — Engine shutdown and free fall. At 13 ± 4 ft the radar puts out a signal which causes engine shut-off and free fall to the lunar surface. This portion of the test is critical primarily with respect to uneven engine shut-off resulting in excessive angular moments being

imparted to the vehicle. The T-2 vehicle in testing this part of the flight will be able to determine accurately the moment disturbance due to engine shutoff. Since the T-2 moment to inertia ratio is larger than for the lunar vehicle, this angular disturbance will be magnified, thus allowing a more accurate determination of engine shut-off dispersions. The final test will result in an actual soft landing and thus will test under dynamic conditions the 13 ft radar mark circuit, and the resulting flight control switching operation.

e. T-4 system test planning. Preparation of a detailed test specification for the T-4 proof-of-design vehicle was begun during this report period. As a basis for defining the required specification inputs, a preliminary T-4 test sequence was prepared (Fig 143). The test program can be logically divided into four phases: system functional operation, environmental, spaceframe coupling, and real time mission on sequence tests. These phases are further divided in the 15 operational modes defined by the *Surveyor* System Functional Requirements Specification. General test procedures for the parallel group tests (test blocks 1 through 16) are 60% complete and integration of these procedures has begun.

System functional operation tests. This phase of testing covers all activity from initial power application through the complete ambient environment mission simulation tests. As part of the spacecraft assembly sequence, unit test access adaptors will be installed between the spacecraft harness and several of the spacecraft units. These adaptors will provide the required electrical isolation between the various test groups during the parallel group tests, and will allow detailed monitoring capability during the integration and system tests. An additional objective during this test phase will be an evaluation of system performance under exposure of simulated *Atlas/Centaur* and RF radiation sources at the Atlantic Missile Range.

Environmental tests. Spacecraft operation during and following exposure to five types of environment will be examined during this test phase. These environmental conditions are heat sterilization, gas sterilization, vibration (boost and descent), thermal-vacuum (transit, lunar day, and lunar night) and shock (drop).

Immediately preceding and following each environmental test phase, the spacecraft will be given a complete system functional checkout in order to isolate any failures to a particular test environment. During the lunar portion of the thermal-vacuum tests, it will be necessary to remove certain units and spaceframe com-

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ponents which are not required to function after touch-down and which would be damaged by exposure to this environment. These include: (1) vernier propulsion components, (2) flight control sensor system components, (3) flight control radar components, and (4) landing gear shock struts. The compartment covers will be replaced with special thermally equivalent test covers which will provide special instrumentation access and the necessary clearance for the unit test adaptors.

Another area under consideration involves determining appropriate methods for simulating the functioning of mechanisms under earth gravity conditions, where the design environment is moon gravity.

Spaceframe coupling tests. These tests, which will be conducted at the Air Force Missile Development Center in New Mexico, will determine if any adverse coupling exists between the vernier propulsion system and the flight control electronics. Since these tests are preceded by the environmental tests, it will be necessary to replace certain components of the vernier system which cannot withstand exposure to the lunar night temperature. As a result, the vernier system acceptance firing, which is normally part of the spacecraft assembly sequence, will not be performed. Instead, the required qualification

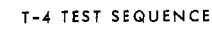
firing will be performed at AFMDC at the start of this test phase.

For the majority of these tests, the spacecraft will be supported at its center of gravity by a special support frame and an adapting fixture which mounts in place of the main retro engine. Weights can then be attached to the support adaptor fixture to simulate the retro engine. With the flight control system operating in the inertial stabilizing mode, the vernier engines will be fired and a disturbance impulse applied to the spaceframe. The gyro outputs will be monitored to determine the effect of any spaceframe coupling. The effect of the support adaptor will be examined by repeating certain tests while the spacecraft is supported from above by elastic shock cord attached to a tower. Only the flight control portion of the spacecraft electronics will be energized during these tests, and all monitoring and commands will be hard-lined.

Real time mission sequence tests. This phase of testing will be divided into ambient environment field tests and thermal-vacuum chamber tests. One important objective of this test phase is to obtain power-profile data as the spacecraft is commanded on a real time basis through all modes of operation. During the field tests, an auxiliary power supply will simulate the output of the solar panel at near lunar noon conditions.

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Figure 143. Test sequence for T-4 test spacecraft

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L. Mission Operations

1. Surveyor Pad Safety Report

A Surveyor Preliminary AMR Pad Safety Report (Report 223450) has been prepared to comply with AFMTC General Safety Plan, Volume I. The purpose of this report is to describe the spacecraft operations at Atlantic Missile Range (AMR) and handling of hazardous items. The potentially hazardous items to personnel and property presented by the shipping, handling, assembly, loading, testing, and sterilizing of the spacecraft are as follows:

- (1) Main retro-rocket.
- (2) The vernier engine system, including fuel and oxidizer.
- (3) Pyrotechnic devices.
- (4) Pressurized helium and nitrogen tanks.
- (5) Radioactive material.
- (6) Toxic sterilants.

A complete safety program must be developed and implemented at AMR for handling the spacecraft because of the unique safety problems resulting from sterilization and final assembly requirements and the radioactive materials.

A presentation of the proposed AMR safety plan was made during the week of February 5 to the NASA

Launch Operations Directorate and to the AMR Range Safety personnel. Informal comments received at these meetings were favorable although some concern was expressed about the need for additional safety measures in the area of the main retro-rocket. It is anticipated that written comments will be received shortly outlining any specific changes that Range Safety may require on the spacecraft.

2. Mobile Sterilization and Temperature Control Unit (MSTCU)

The procurement specification for the MSTCU is in final form and will be released for bidding. The design and development will begin in March, 1962. It will perform terminal surface sterilization and temperature control of the spacecraft while sealed within the GDA nose fairing.

The MSTCU will introduce into the sealed nose fairing a sterilant gas supply mixture of 12% (by weight) ethylene oxide (ETO), and 88% (by weight) Freon₁₂ gases, blended at a relative humidity of $35 \pm 15\%$. The MSTCU simultaneously will control the temperature within the nose fairing during the sterilization process in a temperature range of 68 ± 2 to $86 \pm 4^\circ\text{F}$. The gas sterilization unit will create a low-pressure vacuum in the nose-fairing cavity prior to introduction of the sterilizing gas mixture.

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A humidity-blending chamber will be provided on the MSTCU and will control the humidity throughout the sterilization process. The MSTCU will regulate and maintain the blended sterilizing gases within the nose-fairing cavity for a period of 10 to 24 hr. Pressure gages and indicators are incorporated on the MSTCU for direct readout of pressures, temperatures, humidity blending, gas concentration, gas quantity, and gas flow during the sterilization process. Blending and gas flow operations are semiautomatic, and provide for final adjustment and overriding of the automatic system.

The MSTCU will have the capability of removing the sterilant gases from the nose fairing cavity and purging with sterile nitrogen gas for a period of 4 hr. The nitrogen will pass through 0.3μ filters to ensure sterility.

A disposal system is incorporated on the MSTCU compatible with the specified sterilant gases, and is capable of transporting varying amounts of waste gases from the nose-fairing cavity to the disposal unit for detoxication and disposal. In the event of emergency, the disposal system is capable of detoxicating and disposing of all gases from within the nose-fairing cavity.

The temperature control unit is attached to the sterilization unit and is capable of operating independently. After the spacecraft is sterilized, the temperature control unit will be detached from the sterilization unit and will operate from a commercial or a mobile power source, depending on the existing conditions. The temperature control unit will maintain the temperature range within the sterilized nose fairing from 68 ± 2 to $86 \pm 4^\circ\text{F}$ continuously during the sterilization process, storage, and transportation of the spacecraft to the launch pad.

The temperature control unit will be capable of automatically regulating the temperature within the nose fairing cavity at the required levels. After turnon, the spacecraft receiver will operate on power of 9 to 100 w, depending upon checkout period duration. During transportation, heat from the Sun will create an additional thermal load of 200 to 400 w. These conditions will create varying thermal levels within the nose-fairing cavity which must be automatically sensed and regulated by the temperature control unit to remain within the specified temperature range.

3. On-Stand Air-Conditioning Unit

The air-conditioning unit will be designed and manufactured by General Dynamics/Astronautics (GDA).

The GDA air-conditioning unit will maintain the sealed, sterilized spacecraft within the specified temperature range during the *Surveyor/Centaur* mating operations until liftoff.

Three special biological filter units will be supplied by Hughes Aircraft Company to GDA (one internal unit, two external units) to maintain the spacecraft in a sterile condition. Sufficient spare filter elements will be supplied to GDA by Hughes Aircraft Company to assure continuous maintenance of the spacecraft in a completely sterile condition throughout the launch operation.

The two external filter units will be installed in the air-conditioning ducting by GDA. Each unit contains replaceable filter elements. Only one element per unit will be utilized at any given time during the cooling operation. When a contamination level of 30 ± 4 psi has been reached, the air flow will be transferred to a clean element without interruption of countdown or cooling operations. After transfer from the contaminated filter element to the clean element, the contaminated element will be replaced.

The internal filter element will be inserted in the inlet duct. The duct shall then be sealed (airtight) into the flexible diaphragm between the *Centaur* and the spacecraft. The duct containing the biological filter shall be routed from the seal to a point near the present *Centaur* electronic cooling air umbilical island in the cylindrical section of the nose fairing. The umbilical connection is located 11 deg off the X-axis at Station 170.07 Quadrant II. The internal filter inlet ducting will remain connected to the spacecraft until nose-fairing ejection.

4. Prelaunch TV System Check

As a final readiness check of the spacecraft, several quick qualitative tests conducted almost entirely by RF link from the mobile command and data handling console are planned after the spacecraft is installed on the *Centaur*. It was decided that because it would easily check such a large portion of the telecommunications system, some type of television test should be included in this final readiness check.

A plan has been formulated by GDA and Hughes Aircraft Company for testing the TV system after the spacecraft has been mated to the *Centaur*. The check is required to provide a go, no-go indication of the functional status of the TV system during the countdown. Three possible methods were considered:

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- (1) Provision of an internal light to illuminate the entire payload compartment under the shroud.
- (2) Provision of five separate lights, one for each camera, which could be used as targets.
- (3) Provision of a 4-in.-diameter window in the sterilization bulkhead to allow illumination of the payload compartment from an external ground equipment light.

Method 2 was rejected because it would entail more weight and additional wiring and, in the event of a failure, it would be difficult to determine whether the light or camera had failed. Method 3 was also rejected because it requires that the check be made before installation of the shroud panels. Method 1 was recommended to be implemented as described below.

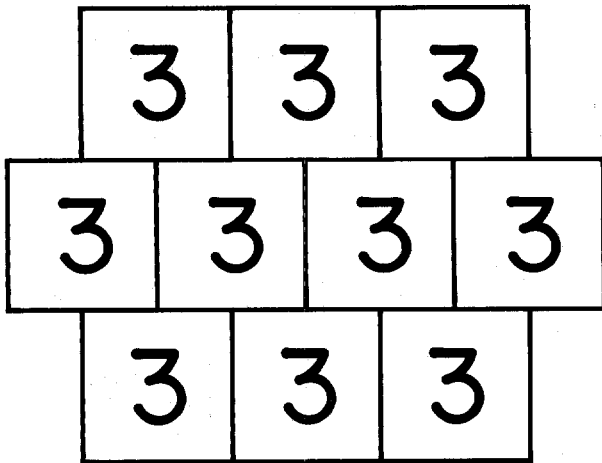


Figure 144. Typical television test target

Two light bulbs of 40w 28v, will be attached to the shroud thrust bottle bracketry which will supply a nominal 8 lu/ft² on the targets. The test targets will consist of patterns of the same number to identify each of the operating cameras (Fig 144). This will allow the shroud to be displaced as much as 1 in. without affecting the test. The test targets will be painted on the inside of the shroud by GDA before gas sterilization at AMR. Target patterns of 9.9 in.² will be required at designated shroud stations (Fig 145). Hughes Aircraft Company will supply the test targets. A hardline switch will be located inside the blockhouse for Hughes Aircraft Company control of the operation of the light. The shroud is being manufactured by General Dynamics, Fort Worth, and is made of white fiberglass so that good reflection inside the shroud is expected. The TV

CAMERA NUMBER	APPROXIMATE DISTANCE TO SHROUD TARGET, in.	APPROXIMATE SHROUD STATION FOR TARGET	ANGULAR POSITION FOR SHROUD TARGET *
1	62	69	-Y + 25 deg
2	62	86	X + 10 deg
3	75	83	-Y + 25 deg
4	41	STERILIZATION DIAPHRAGM 34 in. FROM Z-AXIS	
5	70		X + 25 deg

* ANGLES ARE MEASURED CLOCKWISE LOOKING DOWN

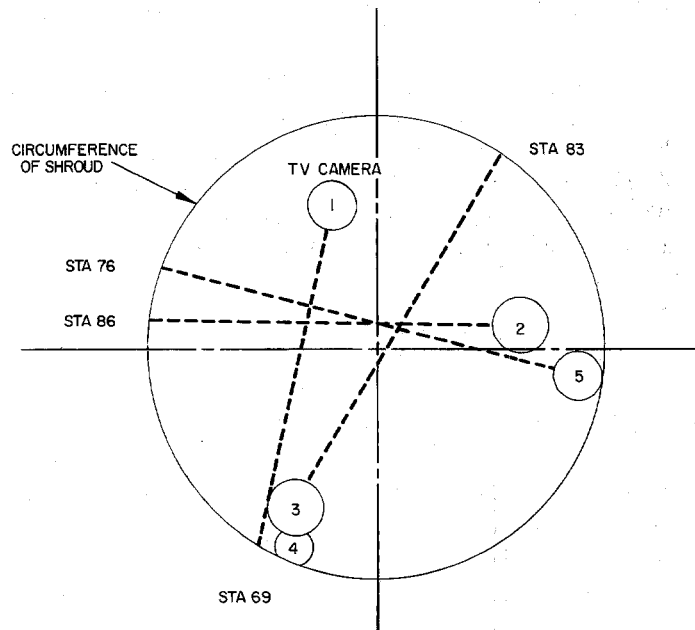


Figure 145. Television target locations on shroud

test will take about 5 min. It is expected that the total weight increase on the shroud from the light source, bracketry, bulbs, and wiring will be about 1 lb. This is approximately 1/15 lb equivalent payload.

5. Compatibility of Spacecraft Radar Microwave Mixers with AMR Radars

During this reporting period, the possibility of damaging the *Surveyor* radar microwave mixers during operations at Cape Canaveral was investigated and is considered unlikely. The allowable electromagnetic power density at the antenna input is a function of the crystal power limitations, the effective receiving cross section of the antenna, and the attenuation in the microwave path from the antenna to the crystal holder. Power density at the antenna aperture is a function of frequency

and angle between the incident power and axis of the main lobe.

Investigation has revealed that electromagnetic power densities at *Surveyor* locations at Cape Canaveral are not sufficient to damage the radar crystals. Also the radar altimeter/doppler velocity sensor (RA/DV) units have a narrow passband and incident frequencies are sharply attenuated on either side. The peak power of a pulse train applied to the HDM-1 crystal (utilized in the RA/DVS) should be limited to 1w to prevent crystal damage. The continuous wave (CW) should be limited to 0.1 w. The RA/DVS waveguide is tuned to 13 kmc. The FPS/16 radar has a frequency of approximately 5.46 kmc; waveguide attenuation at this frequency is roughly 400 db.

Protection of the altitude marking radar is based on the fact that this radar is continuously pointed toward the ground. Since the power density is a function of angle of incidence, it would be extremely difficult to direct incident damaging power. Cape Canaveral operating procedures implement the requirement of pointing toward the ground.

The effective radiated power is listed in Table 27 for two powerful radars at Cape Canaveral; these radars present the most radiation saturation at the sites selected.

Table 27. Effective radiated power

Site: Pad 36

Radiation source	Effective radiated power, mw	Distance, m	Absorbed power, mw
AN/FPQ-4 radar	77.5	9378	1.46
AN/FPS-16 radar	28.2	3855	3.30

Site: Hangars E and F
(Closest hangars to radar source yet still no danger)

AN/FPQ-4 radar	77.5	8528	1.76
AN/FPS-16 radar	28.2	382	336.46

Site: Hangars N and U
(Roughly near proposed *Surveyor* hangar)

AN/FPQ-4 radar	77.5	9493	1.42
AN/FPS-16 radar	28.2	1512	21.45

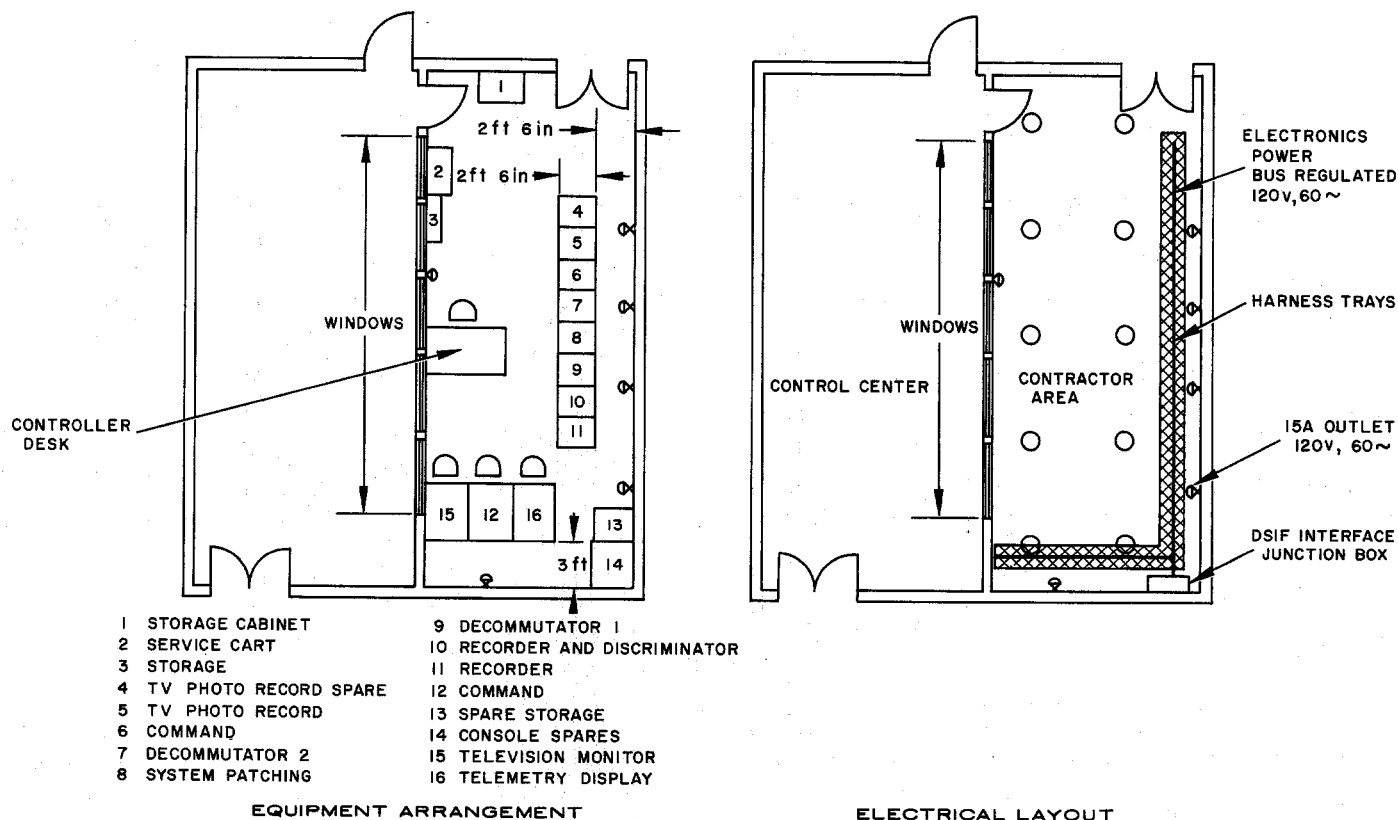


Figure 146. CDC handling facility — Goldstone

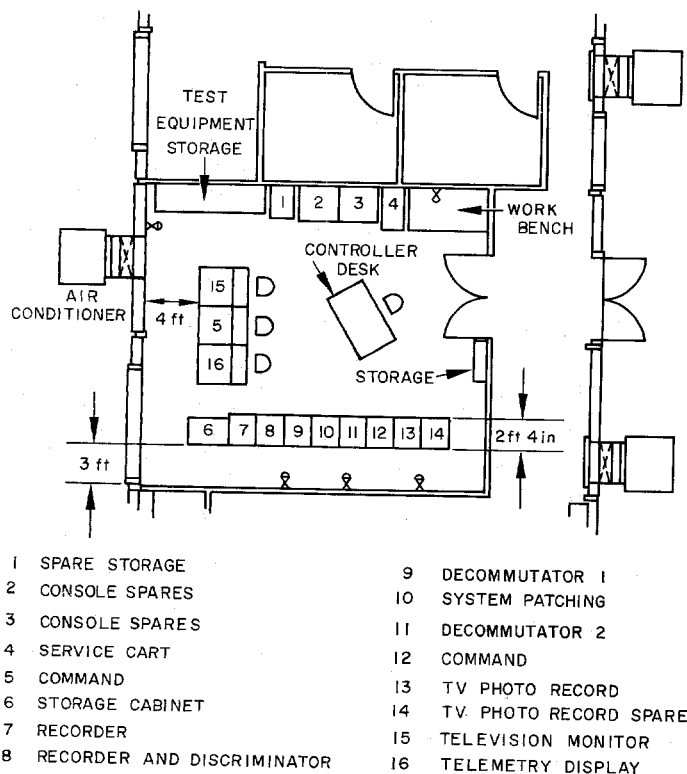
6. Integration of Command and Data Handling Console with Deep Space Instrumentation Facility

Equipment arrangements and electrical layouts for the Hughes Aircraft Company command and data handling facilities in Goldstone, California; Woomera, Australia; and Johannesburg, South Africa are depicted in Figures 146, 147, and 148. In each case, the eight basic racks of CDC auxiliary equipment are to the left and at right angles to the operating consoles. Specific equipment in these racks is shown in Figure 149. The oscillographic recorders have been placed in separate racks for ease of readout. The demodulator, with spare, has been placed adjacent to the operating console to facilitate and expedite emergency changeover. Similarly, storage drawers have been conveniently provided. Office space and maintenance areas have not yet been specified at Goldstone, although sufficient office space is apparently readily available. A calibration-maintenance area and storage space remain to be specified at Johannesburg, where space in general is apparently at a premium.

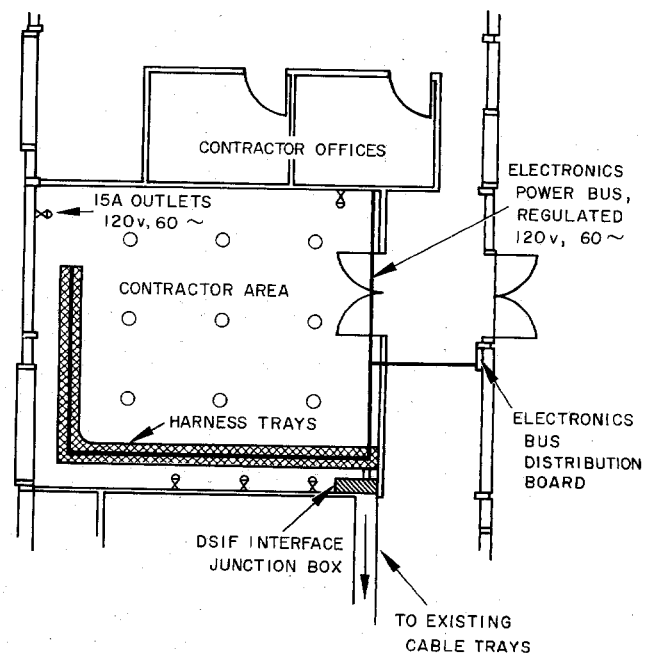
Each CDC system shipped from Hughes Aircraft Company will be identical for installation and standardization purposes. Harness trays will be provided at all sites, as indicated, to provide flexibility and ease of installation.

The signal interface between CDC and Deep Space Station (DSS), through DSIF-installed junction boxes, is shown in Figure 150. As can be seen, the DSIF will provide a video tape recorder for use during TV sequences and a narrow-band tape recorder for use at other times during the *Surveyor* mission. The intercom "hot-line," SCO output, and time, are recorded on separate tracks of the narrow-band recorder and are additionally multiplexed on one track of the video recorder. The DSIF will demultiplex this track for playback as required locally by Hughes Aircraft Company.

The Hughes Aircraft Company photo recorders serve as backup to the video recorder while the video recorder serves as backup to the narrow-band recorder when tapes or heads are being changed. When film is employed, time is also photo-recorded from a tuning fork clock face. High-frequency oscillographic backup will be



EQUIPMENT ARRANGEMENT



ELECTRICAL LAYOUT

Figure 147. CDC handling facility — Woomera

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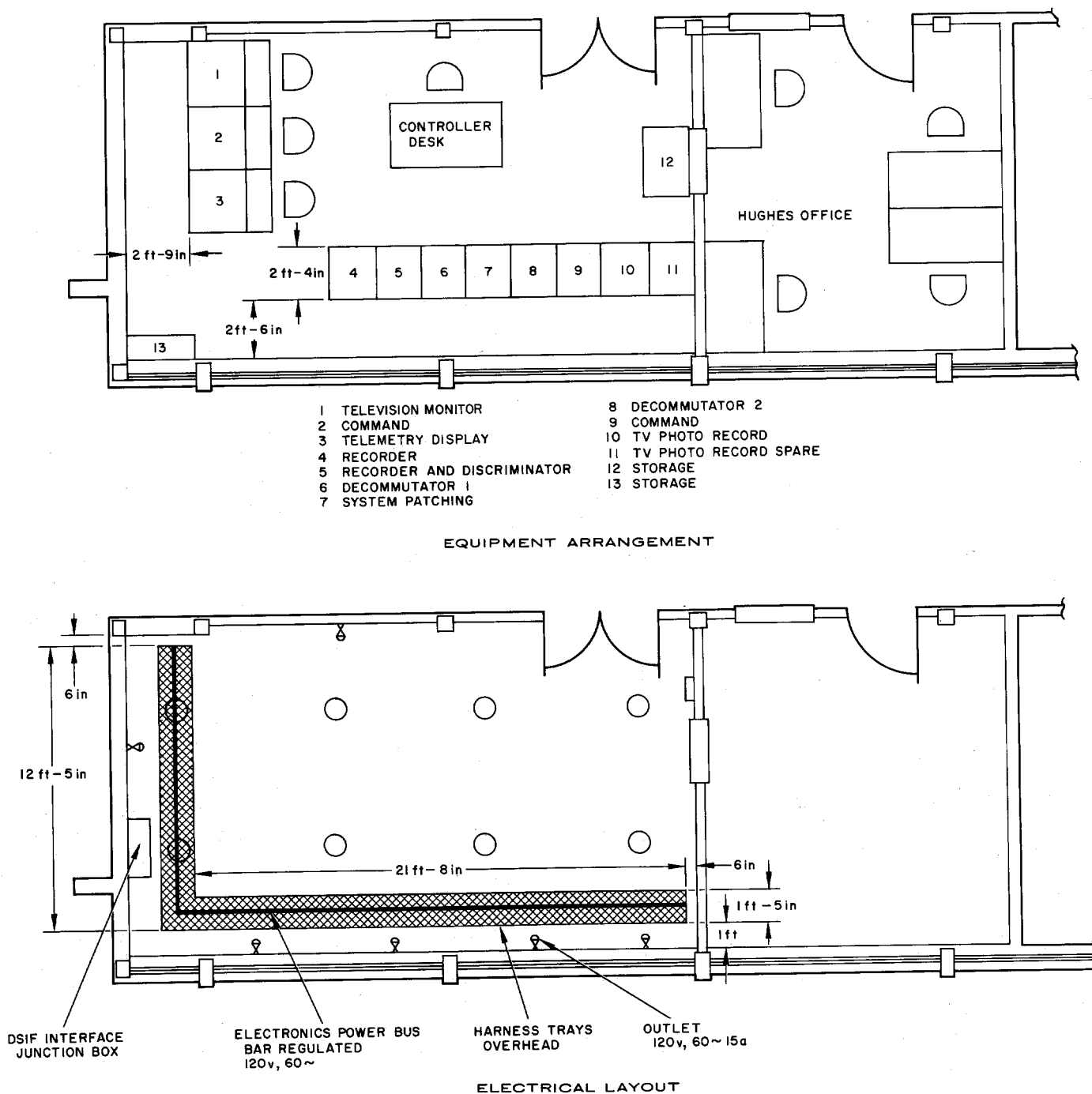


Figure 148. CDC handling facility - Johannesburg

provided by the DSIF at all Deep Space Stations. Low-frequency oscillograph backup will be provided only at Goldstone.

The overseas DSIF stations will provide telemetry-to-teletype encoders for the transmission of selected decommutated data in real time to the DOC as com-

munications permit.

7. CDC Development

a. Status summary. Six of the 12 subcontract items for the prototype CDC have been received from the vendors. The last one is scheduled to arrive March 21,

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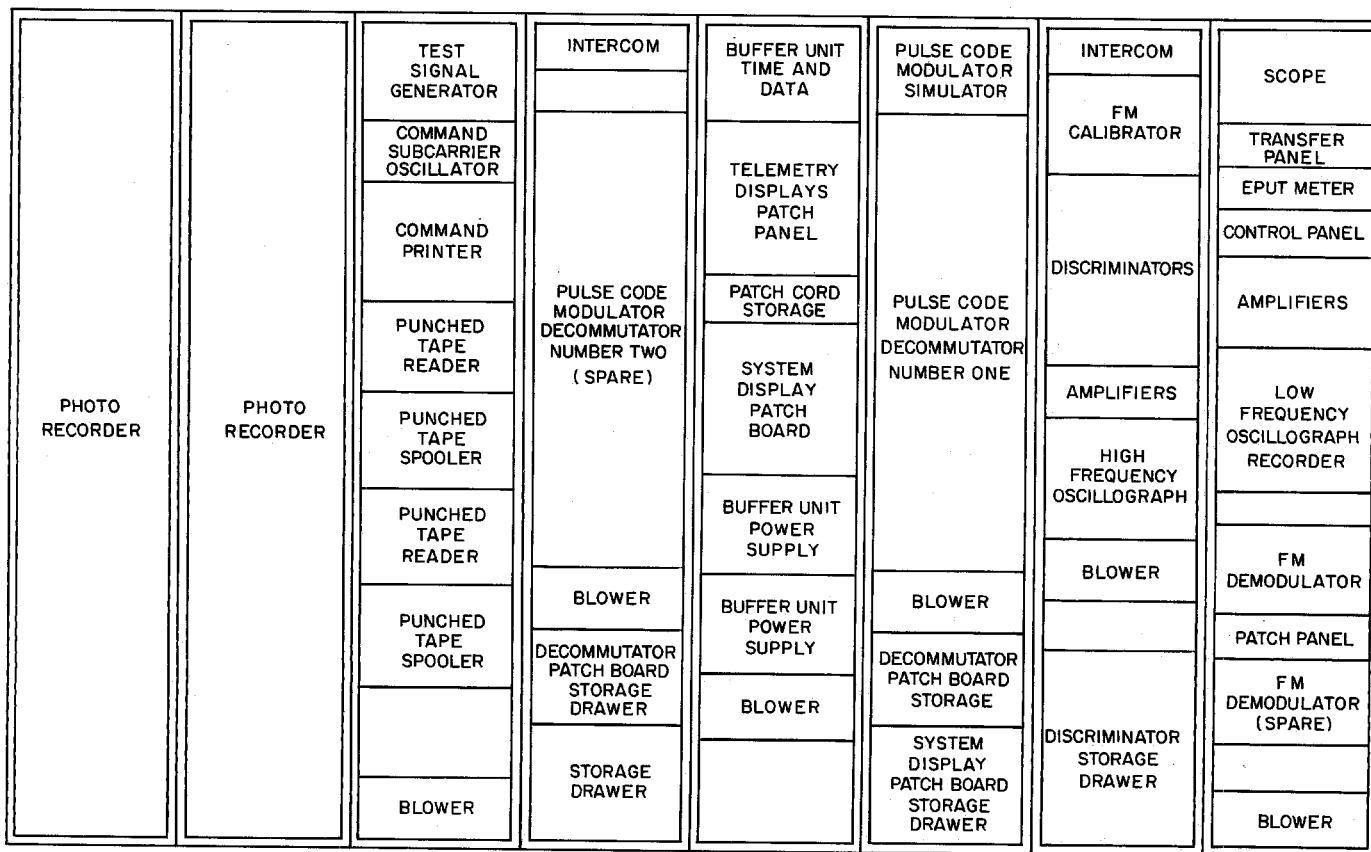


Figure 149. Arrangement of CDC racks at DSIF stations

1962. The subcarrier oscillator assembly has been completed and checked out. All other Hughes-fabricated prototype units are in process of being wired. Racks are being wired and interconnecting cables are being fabricated. Design reviews have been held on the television subsystem, the telemetry subsystem, and the command subsystem of the CDC. These design reviews covered all the design activities leading up to prototype fabrication.

b. CDC command enable/reject modification. A study of various methods for verifying spacecraft reception of CDC command messages has resulted in a change to the original concept. The original method visualized a message reject signal from the spacecraft, which was to be displayed at the command console, when the decoder failed to properly complement check a command word. The plan now is to transmit either command reject or enable signals from the spacecraft decoder. In order to distinguish between bonafide commands and fill-in words, the fill-in words will purposely be programmed with wrong complement bits. Thus, all fill-in

words should be rejected by the spacecraft decoder and a reject signal transmitted, whereas all valid commands should be accepted and an enable signal sent. The reception of reject signals on fill-in words and enable signals on commands then, is positive indication that the spacecraft decoder and the rest of the link are operating properly. The advantages of this design concept are numerous. In addition to making a clear distinction between fill-in words and actual command words, it is of great use as an operational aid to the command operator. Isolation of command link failures becomes considerably easier and less time consuming. Transmission of the command enable signal gives the CDC Command operator a positive indication that the last command word was properly processed by the spacecraft decoder. Transmission of reject signals on fill-in words gives the command operator a continuous general check on the command link. (Failures resulting in marginal signal-to-noise conditions would probably not be distinctively indicated, however, since the resulting bit errors would result only in normal decoder reject signals.)

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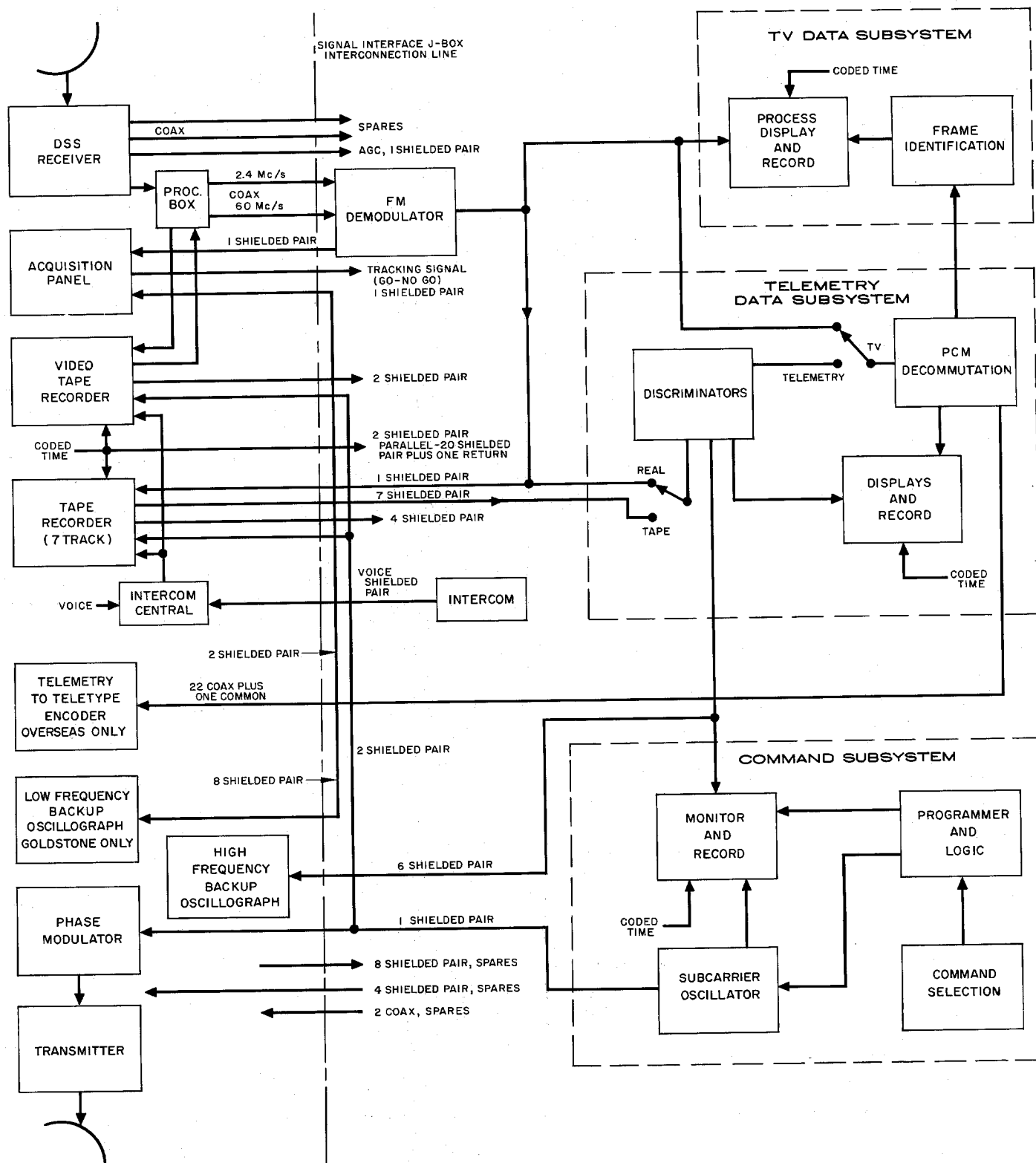


Figure 150. Signal interface, CDC-Deep Space Stations

A change has been made in the method of transmission of reject and enable signals. Originally the message reject signal was to be sent via a separate subcarrier oscillator. However, it is now planned to send these signals as two bits of a PCM word on all engineering commutators. Since the change will not include scientific commutators, the capability for sending reject and enable signals by subcarrier oscillator will be maintained. During those periods when the necessary bandwidth is available, this subcarrier oscillator can be turned on by command. The reject/enable signals will take the form of center frequency to bandedge steps of subcarrier oscillator frequency for 20 ms periods.

The command console display of these signals will be mechanized as follows. Reject signals will be displayed on a "link operating" indicator. This light will be held on for a period of 0.5 sec after reception of each reject signal by the subcarrier oscillator or held on by the decommutator memory for one PCM frame after reception of a PCM reject signal by the decommutator. Command enable signals, whether transmitted by separate subcarrier oscillator or by PCM, will turn on a "command enable" indicator on the command console. This indicator light will be held on until the next command word (not fill-in) is transmitted by the command generator.

c. CDC television/command synchronization. Details have been developed in the design of the CDC com-

mand system for synchronization of the taped program TV sequence commands with the spacecraft TV system. It is necessary that each TV picture transmission and mirror movement be commanded within the time constraints as illustrated in Figure 151. Figure 151a shows the normal television timing with vertical blanking pulses I, II, and III, the video data period, and the two erase periods. Transmission occurs from the beginning of pulse I to the end of pulse II with frame identification data being sent during pulse II. Figure 151b shows the mirror action (0.1 sec for movement and a maximum of 1.5 sec for settling). Mirror action must occur between the end of pulse II and the beginning of the next pulse I. Figure 151c shows the time periods A and B when mirror commands and start picture commands may be accepted by the television system. Acceptance of picture commands during period B causes the leading edge of blanking pulse I to open the camera shutter and also start transmission of pulse I, video, and frame identification during pulse II. Since the nominal TV period of 3.6 sec may vary plus or minus 10%, commands must be initiated such that they occur within the allowed periods A and B of Figure 151c. The method of synchronization adopted conforms to the following constraints:

- (1) The television period must have excellent short-term stability, but the long-term stability may be $\pm 10\%$ (with the nominal at 3.6 sec).

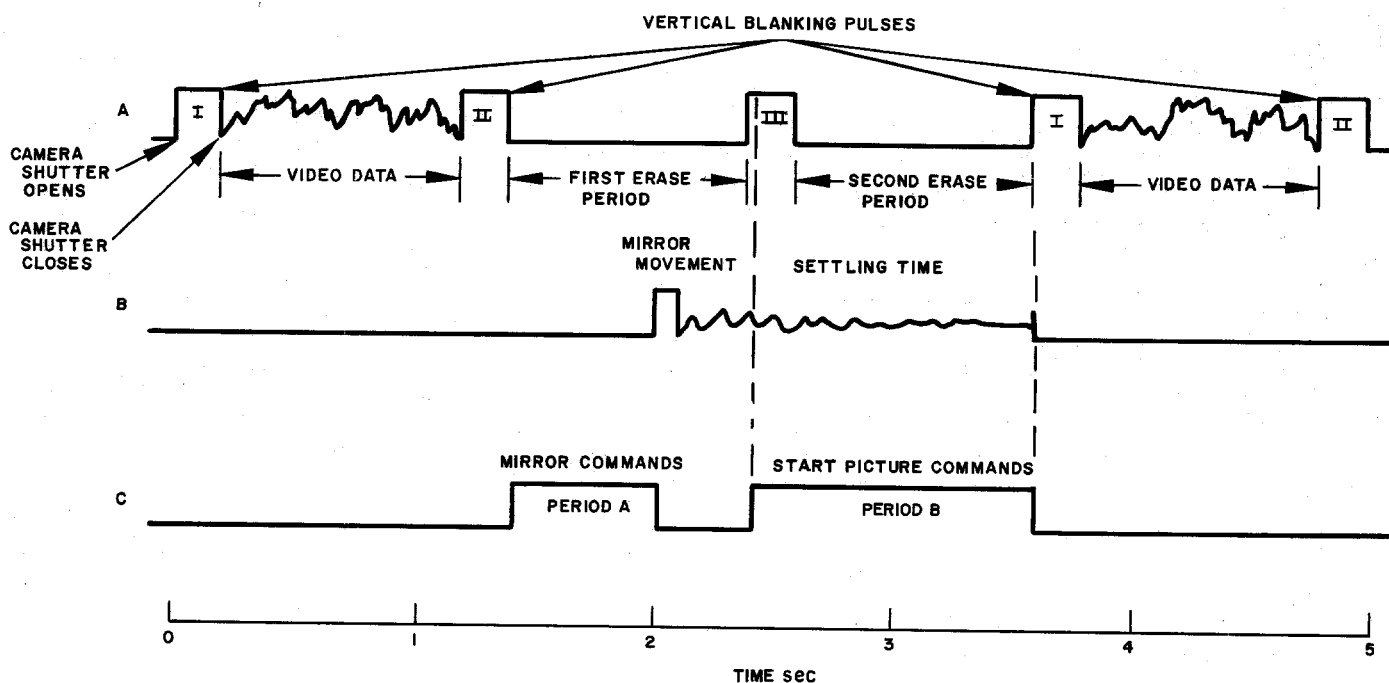


Figure 151. Television command timing restraints

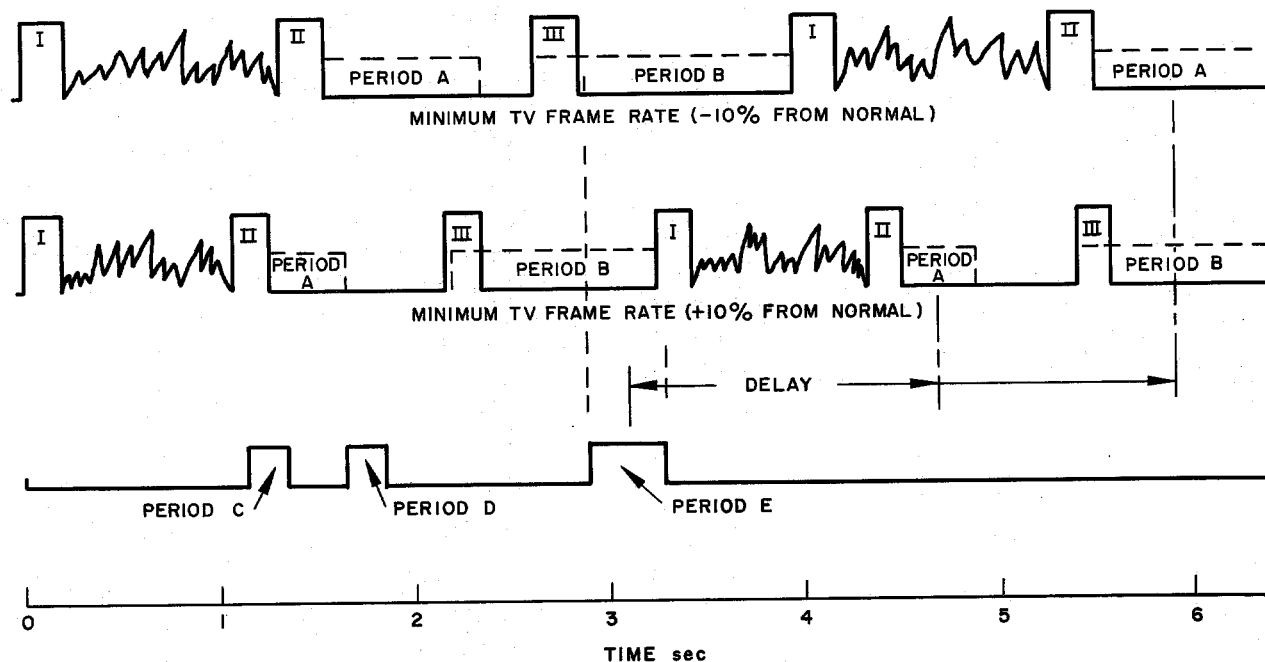


Figure 152. Timing of television commands

- (2) No start picture commands may arrive out of the decoder except during the period from just after the leading edge of blanking pulse III to coincidence with the leading edge of blanking pulse I (period B in Figure 151c).
- (3) The period allowed for mirror movement and settling will be 1.6 sec maximum, which is based on 0.1 sec movement and 1.5 sec settling. The method of mirror command control decided on must also be satisfactory if less than 1.5 sec is required for settling.
- (4) No commands of any kind may arrive out of the decoder during data transmission.
- (5) The RF transmission time from the spacecraft to Earth will be 1.25 ± 0.10 sec with the variation, which is a result of all expected conditions, being very slow.
- (6) The command word length must be 0.5 sec.
- (7) The time for a TV survey must be kept to a minimum in order to obtain as much total TV coverage as possible during the mission.

The command half-second periods are phased in response to the reception of each vertical blanking pulse I

from the preceding picture. This is accomplished without affecting the spacecraft decoder by continuously transmitting fill-in bits after completion of the preceding command word until the rephasing is accomplished, the next bit then becoming the first bit of the 0.5 sec command word.

Shown in Figure 152 are the cases of minimum and maximum TV frame rates with zero time shown as the beginning of the vertical blanking pulse I, following a picture command in an immediately preceding period B not shown. Initial picture commands establish the first period B, which may start just after the leading edge of any vertical blanking pulse.

The third line of the figure shows certain timing relations in the commands. Period C is the interval when the leading edge of pulse I arrives on Earth, taking into account the variation in RF transmission time. On receipt of pulse I a start frame command will be transmitted to the spacecraft. Period D represents the time of the end of the command and is simply period C displaced $\frac{1}{2}$ sec plus 0.02 sec maximum for synchronizing with the next fill-in bit which then becomes the first bit of the 0.5 sec command word.

Period E represents the interval over which the start frame pulse is delivered out of the command decoder to

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the TV system on the spacecraft. Its width is twice the period C width to account for the round trip RF transmission time variation and it is displaced an additional 0.08 sec for synchronizing at the spacecraft to the 4 sync bits in the command.

A delay time is shown which begins during period E and terminates at a variable position so that it will fall during period A. This period represents the delay between the start frame command and the mirror movement command. The delay will be established in the command generator by a manual adjustment based on the earth-to-spacecraft distance and a measurement of the TV frame rate.

As shown in Figure 152, the start frame command can occur somewhat after the leading edge of blanking pulse I under a combination of long RF transmission time and high TV frame rate. To avoid this undesirable circumstance, the spacecraft maximum frame rate is limited to plus 8% instead of plus 10%.

Adjustment of the command generator for the proper delay between the start frame command and the mirror movement commands will be performed manually by the CDC crew at the DSIF stations. A measurement of the time interval between receipt of two consecutive vertical blanking pulses I will be made and a determination of Earth-to-spacecraft RF transmission time will be made either by measurement or from prepared tables. These two values will be entered into a simple equation to arrive at a computed value for the proper setting on the command generator. It is expected that an adjustment will be needed only four times an hour at the most and usually only once an hour.

d. Demodulator. The CDC demodulator accepts IF signals from the DSIF receiver and recovers the video signals generated in the spacecraft. The demodulator operates in one of four modes: (1) normal television and wide-band telemetry, (2) phase modulation, (3) narrow-band telemetry, and (4) emergency television.

In the first mode, a wide-band frequency discriminator demodulates the signal. The second mode is used during transponder operation when the DSIF receiver is tracking the received signal. In this mode a phase tracking loop in the demodulator also tracks the received carrier and the video is obtained from the output of the phase detector. The other two modes differ only in bandwidth. The IF signal is obtained from the narrow-band output of the DSIF receiver and is demodulated with a frequency discriminator. For emergency television, a crystal

filter is inserted in the circuit to provide further narrowing of the predetection bandwidth.

The design of the wide-band portion of the CDC demodulator has been released for product engineering and fabrication. System tests and threshold measurements are being made and will be used in testing of the final units. Design of the 2.4 mc IF circuitry, wide-band video filter, and emergency mode video filter has been completed. A design review of this circuitry will be scheduled soon and the circuit will then be released for product engineering.

Wide-band system video filter. The wide-band video filter is a passive network, consisting of two $m = 0.6$ terminating half sections and two constant-k full sections, driven by an emitter follower with 3 db insertion loss. This low pass filter improves the signal-to-noise ratio of the wide-band system by 10 db. A schematic is given in Figure 153 and the transfer characteristic is shown by Figure 154. The transfer function is down 3 db at 220 kc and has a slope of approximately 450 db per octave.

Emergency mode video filter. The emergency mode video filter is an active network of unique design, composed of two independent second-order sections, each built around a transistor gain stage. This low pass filter provides a voltage gain continuously variable from 0.5 to 23.5, attenuation of -3 db at 1000 cps, and a -24 db per octave cutoff slope. The maximum possible error in corner frequency, due to limit transistors, is $\pm 4\%$. Bias temperature compensation is accomplished with two diodes and a sensistor. A schematic is given in Figure 155 and the maximum gain transfer function is shown in Figure 156.

2.4-mc IF circuitry. A schematic diagram of the 2.4 mc IF circuitry is shown in Figure 157. The circuits consist of a tuned cascode amplifier stage followed by a resistive pad and a crystal filter. Either the pad or crystal filter is switched into the circuit by the two relays. In the narrow-band telemetry mode of operation, the resistive pad is in the circuit. In the emergency television mode of operation, the crystal filter is in the circuit. The amplifier output is amplitude-detected, filtered, and fed back to control the gain of the cascode amplifier. The amplifier output is also limited and applied to a frequency discriminator circuit. The frequency discriminator has a frequency peak-to-peak separation of 75 kc and a voltage sensitivity of 0.05 v/kc. The discriminator output is supplied to the emergency television mode filter and the summing amplifier.

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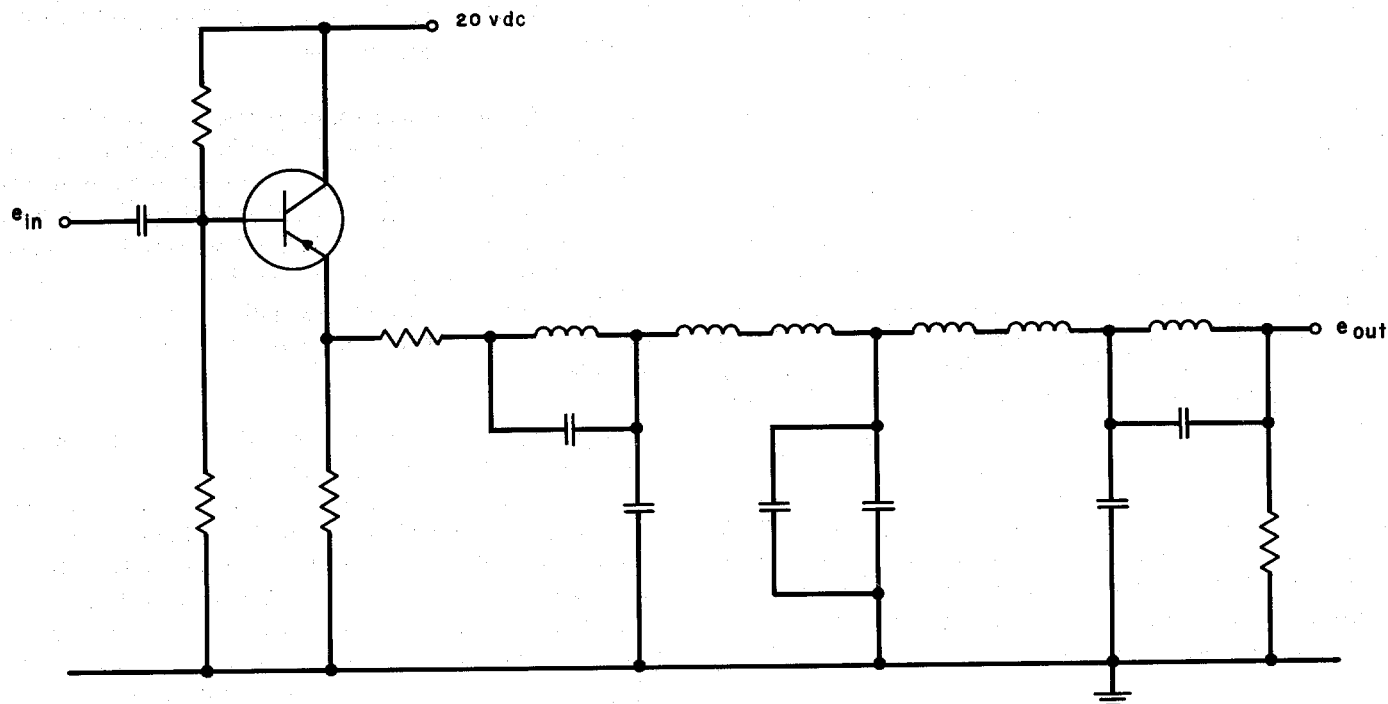


Figure 153. Low-pass, sharp cut-off filter for wide-band FM channel in CDC demodulator

8. Installation, Checkout, and Pre-mission Operation of CDC and SSS at DSIF

A detail schedule for test, installation, and checkout of the CDC and the *Surveyor* spacecraft simulator (SSS)

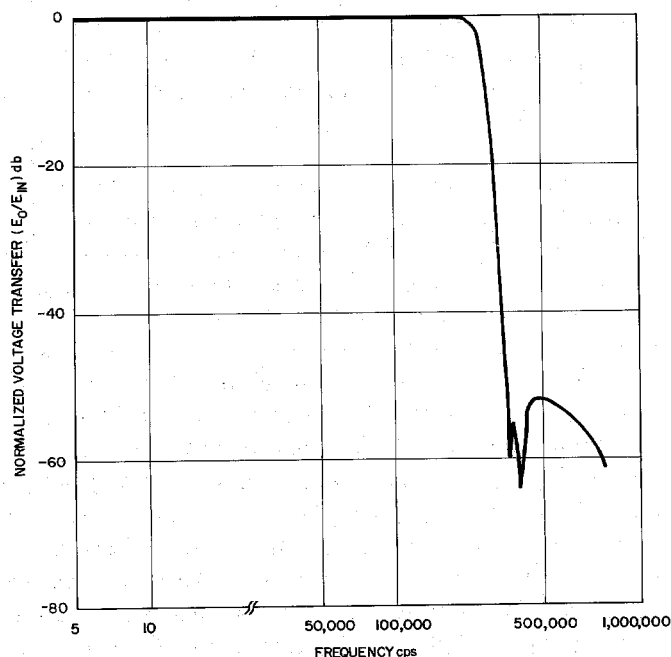


Figure 154. Video filter characteristic

into the DSIF has been developed. The CDC will undergo sequential type approval test stages as follows:

- (1) Individual subsystem tests.
- (2) Subsystem integration tests.
- (3) Spacecraft compatibility tests.
- (4) Operational tests (CDC-SSS loop testing).
- (5) DSIF loop tests (CDC-DSIF-SSS and T4 loop testing).

Detail requirements for Stages 1-4 are being developed. Stage 5 is discussed below with reference to DSIF activity.

This final CDC equipment type approval testing will be accomplished at the Goldstone DSS with the following goals:

- (1) Installation procedure checkout.
- (2) DSIF-CDC interface tie-in and checkout.
- (3) CDC-DSIF-SSS operational testing.
- (4) CDC-DSIF-T4 operational testing.

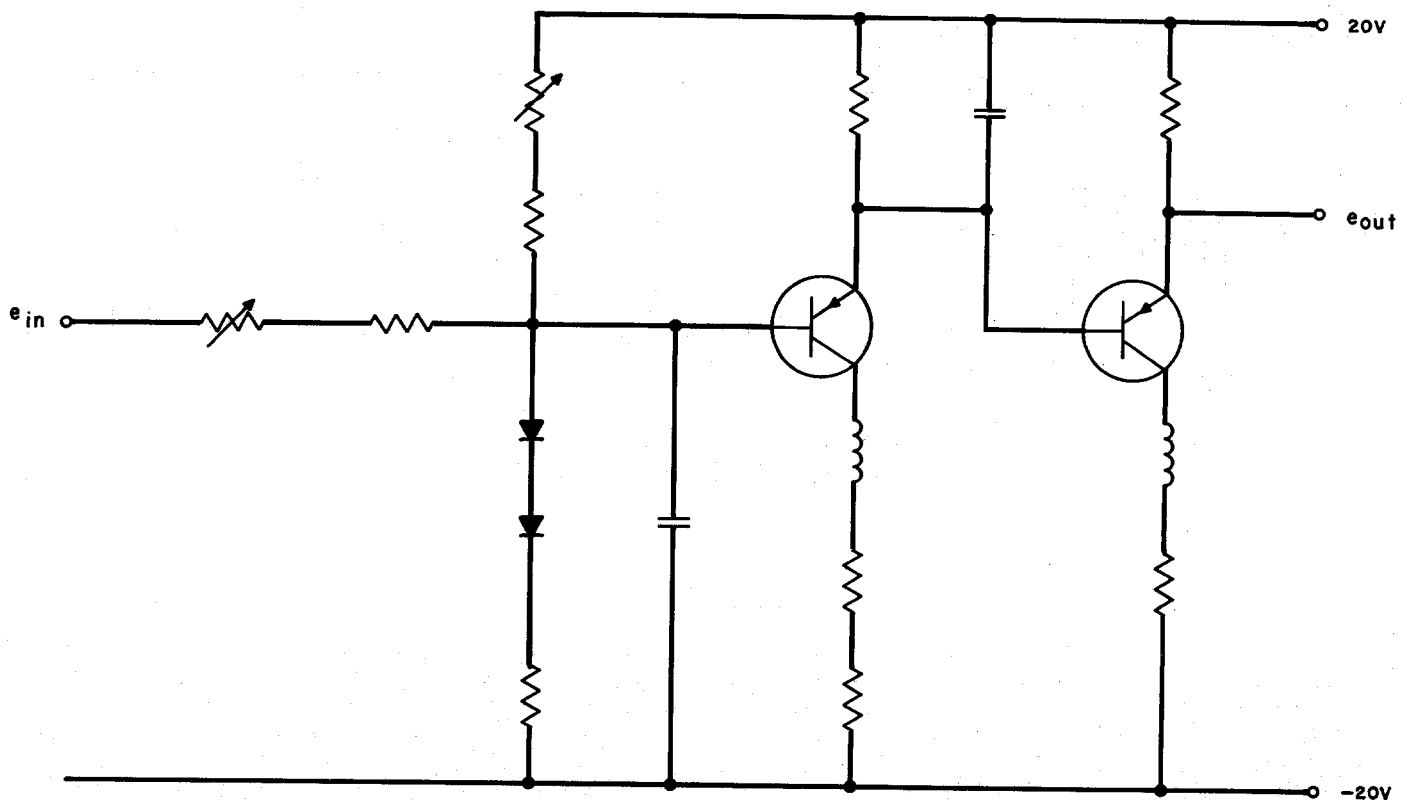


Figure 155. Emergency mode video filter

The Goldstone CDC and spares only will be type tested at Goldstone. The Goldstone CDC will be the first unit assembled and installed and consequently will be regarded as the design control unit. Design changes determined necessary for this unit will be incorporated into the other CDC units at Culver City prior to shipment overseas.

This approach provides for the following advantages over the original plan to checkout each DSIF CDC at Goldstone:

- (1) A more complete CDC checkout will be possible.
- (2) Overseas CDC operating time will be reduced and reliability increased.
- (3) The Goldstone CDC will have a high confidence level.
- (4) Testing costs will be reduced.

Any incompatibilities discovered during this phase will be evaluated and breadboard-type solutions designed and tested at the site. After verification of the solutions, units requiring appreciable modification will be returned

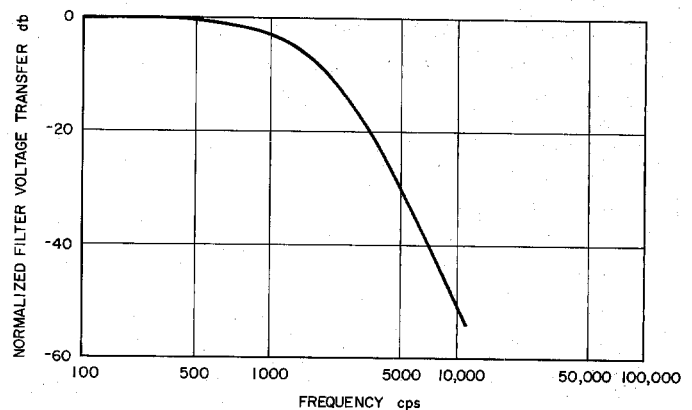


Figure 156. Emergency mode video filter characteristic

to Culver City for rework. Detail procedures for this test stage have been developed and the DSIF time and equipment requirements established. Figure 158 illustrates the over-all schedule and major milestones. Goldstone (GS) DSIF noninterference areas are indicated and building modification milestones are presented. DSIF phases are indicated by number. Individual DSIF phases are defined and summarized in Table 28.

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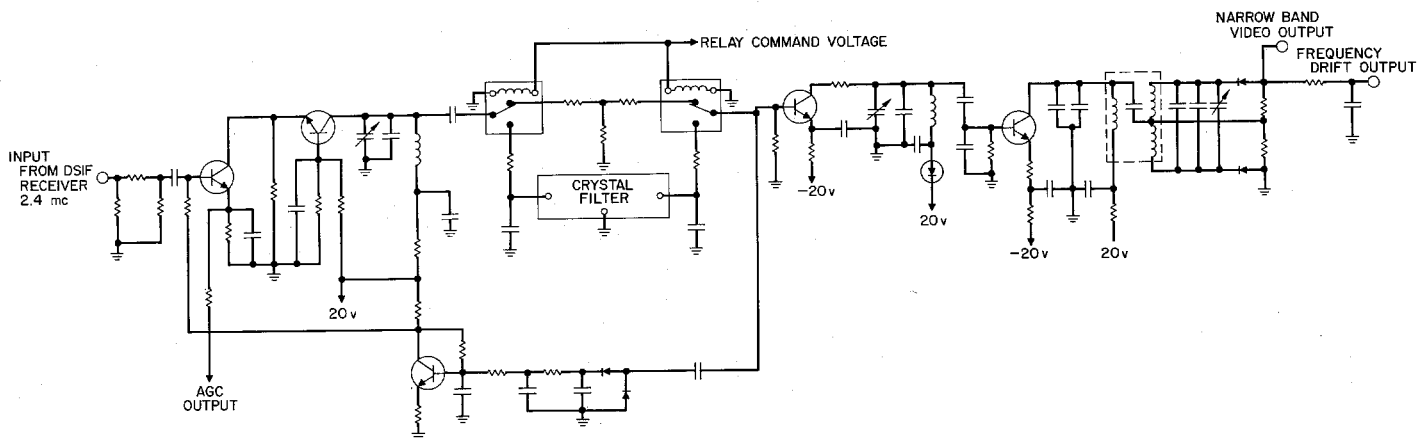


Figure 157. 2.4-mc IF circuitry

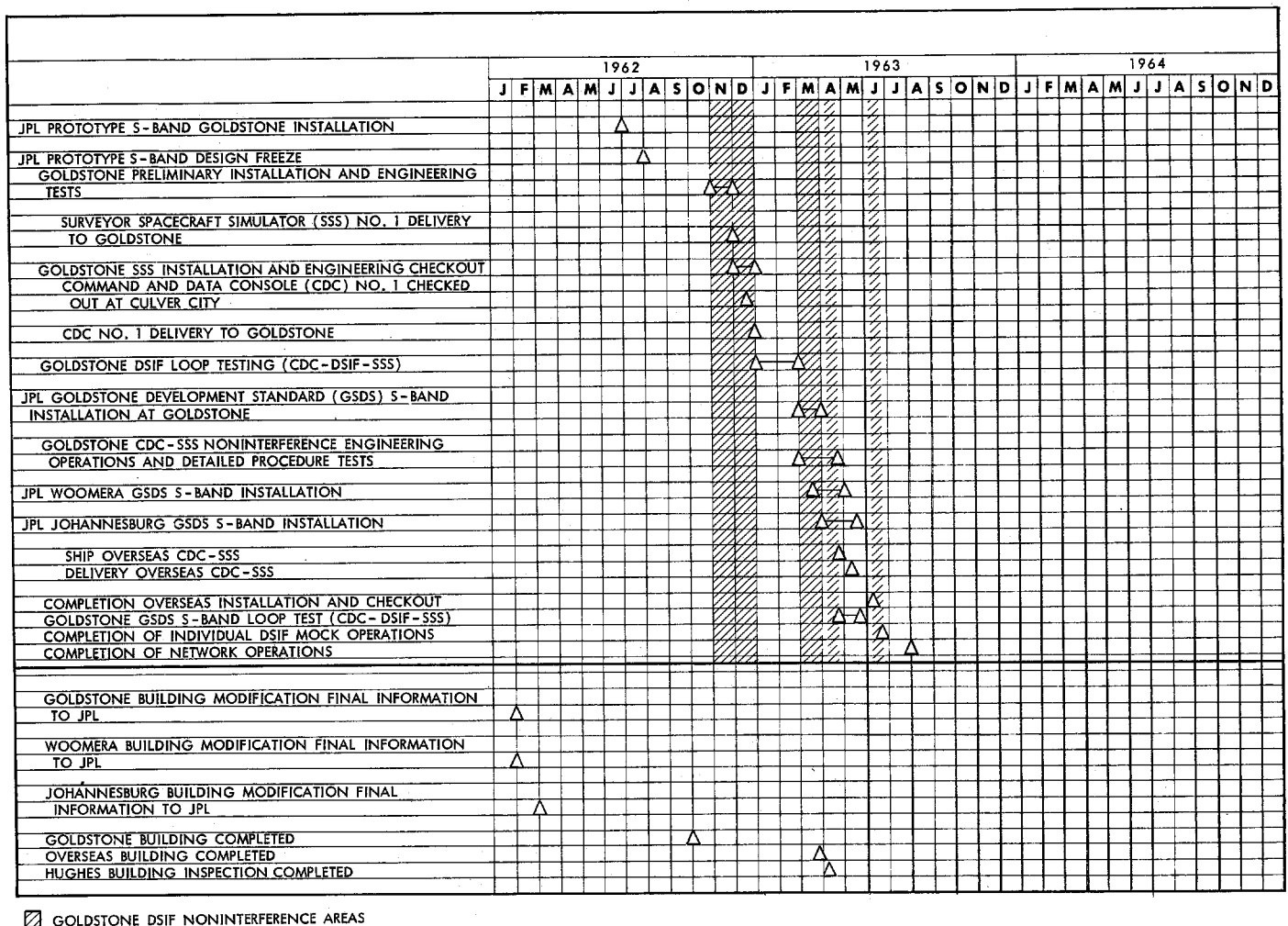


Figure 158. Surveyor DSIF installation and test schedule

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Table 28. CDC-SSS test, installation, and checkout phases

Phase	Time and date	Description
1. GS preliminary installation and engineering test.	1 mo 1 November- 1 December 1962	Pre-CDC-SSS delivery. Noninterference except on selected days as permitted. Install cabling, interface measurements, engineering familiarization and test.
2. GS SSS installation and checkout.	1 mo 3 December- 31 December 1962	Pre-CDC delivery. Noninterference except on selected days as permitted. Installation and subsystem checkout of SSS.
3. GS DSIF loop testing (CDC-DSIF-SSS) including CDC installation and checkout.	9 wk 26 December 1962- 27 February 1963	(a) CDC installation and subsystem checks. (b) Interface tiein and checkout. (c) CDC-DSIF-SSS operational testing.
4. GS CDC-SSS noninterference engineering operational test and procedures checkout	7 wk 1 March- 22 April 1963	Not presently detailed since entirely noninterference. Consists of: (a) Detailed operational procedure C/O. (b) Operator training and familiarization. (c) DOC tiein procedure C/O. (d) Engineering equipment modification evaluation.

Phase	Time and date	Description
5. Overseas CDC-SSS installation and checkout.	1 mo 6 May- 7 June 1963	Same schedule for Woomera and Johannesburg.
6. GSDS* S-band loop test (CDC-DSIF-SSS).	1 mo 23 April- 20 May 1963	Required for verification of GSDS S-band CDC-SSS compatibility. Also includes CDC-DSIF-T4 testing and mission simulation.
7. Individual station mock operation.	2 wk 10 June- 21 June 1963	With all equipment checked out and ready, this phase emphasizes final procedural evaluation and complete station readiness.
8. Full net operation.	6 wk 24 June- 1 August 1963	Final test and procedure checkout. Stations readiness verified with emphasis on data handling, operational control, communication checkout, and DOC optional interface. Mission will be simulated in detail including nonstandard cases.

*Goldstone duplicate standard.

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V. Lightweight Surveyor Study

Hughes Aircraft Company was requested in early January 1962, to perform a detailed study and establish a development plan for a lightweight *Surveyor* spacecraft for possible use during the early part of the Project. The lightweight design was to be accomplished by modifications in the existing *Surveyor* design to reduce the weight of the scientific payload and the main retro propulsion system.

Basic guidelines of the study were that the spacecraft gross separated weight should be approximately 2100 lb., and that the requirements set forth in the JPL specification for *Surveyor* spacecraft system design would apply.

Mission objectives of the lightweight *Surveyor* spacecraft were established as follows:

- (1) To maximize the probability of a successful soft landing on the lunar surface, verify achievement of the soft landing, and supply data to evaluate the engineering performance of the spacecraft.
- (2) Within the limitations of payload constraints, to determine the lunar environment and selenographic characteristics.
- (3) To maximize the probability of lunar night survival.

A. Configuration

One of the primary factors influencing the configuration of the lightweight spacecraft was that of determining an acceptable cg location. The payload had to be so located as to bring the lateral cg of the complete spacecraft as near to roll z axis as possible, while at the same time maintaining compatibility with the stated ground rules and objectives.

When one 45-lb battery and all of the scientific instruments and their auxiliaries are removed from the space-

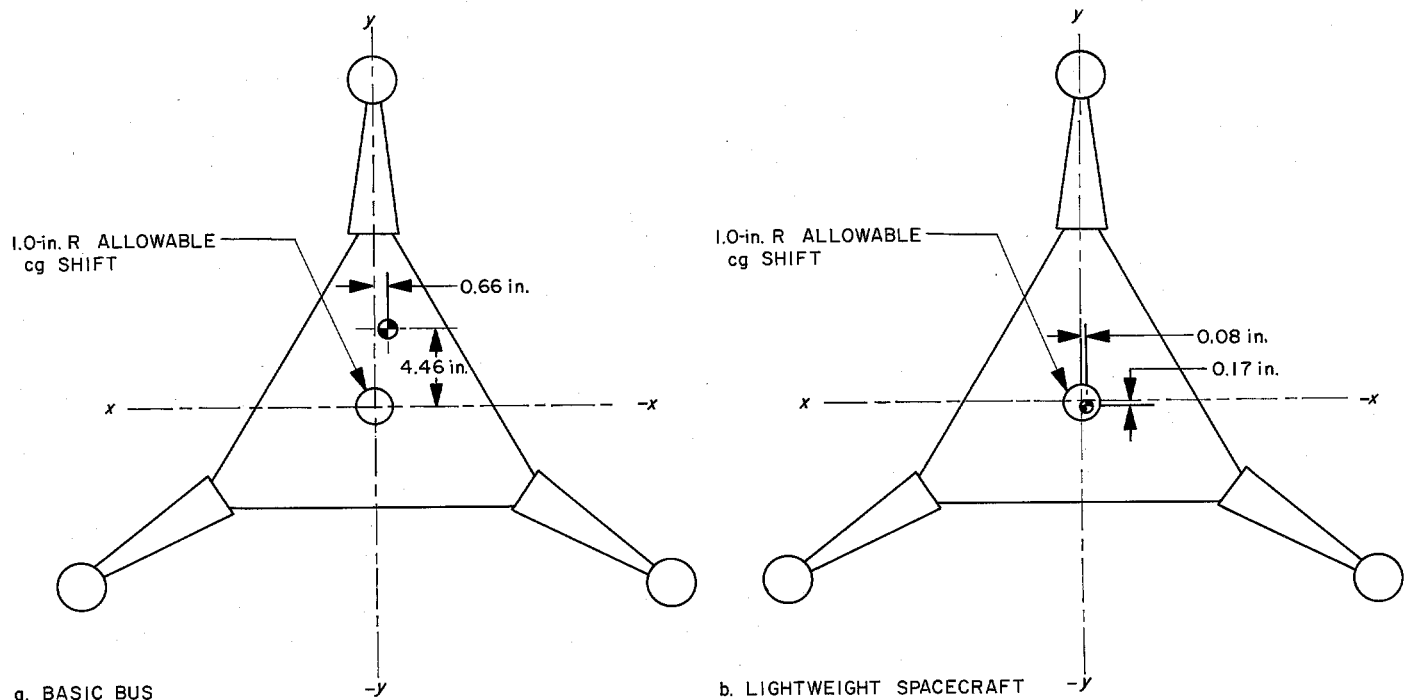


Figure 1. Center-of-gravity locations

Table 1. Payload of lightweight spacecraft

Item	Weight, lb
Engineering instrumentation	10.0
Tape recorder and auxiliaries	10.6
Battery, special	5.0
Compartment D	4.0
Harness	4.0
Television (two cameras with color)	30.6
Television auxiliaries	1.3
Subsurface sampler and auxiliaries	39.5
Subsurface geophysical probe and auxiliaries	4.5
Surface temperature (interferometer spectrometer) and auxiliaries	2.2
Scientific thermal auxiliaries	5.3
Scientific mechanism auxiliaries	2.0
Anchors	10.9
Total payload	129.9

craft, the resultant cg of the basic bus is as shown in Figure 1a. In order to approach the problems of configuring the spacecraft so that the desired cg location could be achieved, it was necessary to carefully select the elements of the payload. The proposed payload elements are listed in Table 1.

The electrical power and mission sequencing requirements were investigated to ascertain whether or not the spacecraft can perform its mission with just the one 45-lb

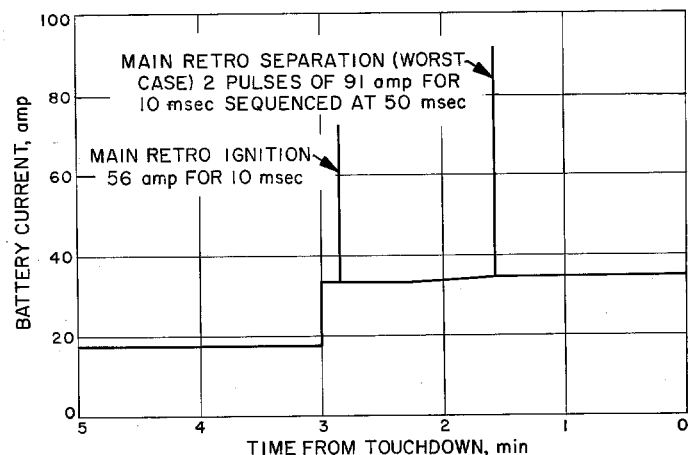


Figure 2. Battery current required during terminal descent

battery carried on the basic bus. It was concluded that the only time power demands will exceed the capability of the 45-lb battery is during terminal descent when the retro engine ignition squibs and the retro engine ejection squibs will be fired. The squib pulses are superimposed on a large steady-state load created by the flight control and terminal descent radar (Fig 2). It was concluded that the problem can be eliminated by using an auxiliary 5-lb battery to fire the squibs; thus a corresponding weight allocation has been added to the payload.

The starting surge current requirements of the subsurface sampler drill motor were also investigated. The drill drive motor will not create a surge current in excess of 60 amp, and the procurement specification is being changed to reflect this maximum allowable surge current. The steady current load, on which this surge is superimposed during lunar operations, will not exceed 5.8 amp. (This load includes TV and high power transmitter requirements.) Thus, the worst case surge during the lunar operations will not exceed 65.8 amp. This is well within the capabilities of the 45-lb battery carried by the basic bus.

Compartment C (used in 2500-lb design) has been eliminated and has been replaced by a Compartment D. Even though there is space available in Compartments A and B for the payload auxiliaries, it is necessary to create a Compartment D in order to achieve the desired location of the spacecraft center of gravity. Compartment D is fundamentally the same as Compartment C, and does not constitute a new development.

The engineering instrumentation listed in the payload summary represents instrumentation in addition to that previously planned for the 2500-lb spacecraft and is consistent with the lightweight vehicle objectives. The tape recorder will be used for engineering purposes during periods of the mission when data cannot be transmitted real time to the DSIF (i.e., boost, injection, touchdown).

The total spacecraft weight is as shown in Table 2 and the vehicle detail weight breakdown and mass property data are listed in Tables 3 and 4. The basic bus weight includes 3.3 lb for planar array-solar panel mast extension, a 45-lb battery and 4 lb of retro engine insulation. The payload weight shown represents a summary of Table 1. Usable propellant weight of 1337.7 lb represents the combined vernier engine and retro engine propellant weight.

The predicted flight weights are lower than the current vehicle design weights and include a number of planned future weight improvements. A continuing program directed toward the reduction of the weight of all parts of the *Surveyor* spacecraft will be carried out; as an example, a more detailed study of the need for the special

Table 2. Summary weight statement

Basic bus	631.7
Usable propellant	1337.7
Payload	129.9
Total weight, lb	2099.3

Table 3. Mass property data

Weight, lb	cg position, in.			Moment of inertia, slug-ft ²		
	z ^a	x ^b	y ^b	I _z	I _x	I _y
Total separated, 2,099.3	59.13	0.00	0.00	203	158	156
Maximum touch-down, 705.2	65.14	-0.04	+0.08	153	96	95
Minimum touch-down, 609.4	66.86	-0.08	-0.17	138	85	84

^a z = spacecraft station; landing gear hinge line: Station 47.48.
^b Retro-rocket lateral cg.

5-lb battery will be made, and the battery removed if not required. Another area where careful design is required to minimize weight is in the spacecraft harness. Weight will therefore be a prime factor in the design decisions for this item. Finally, the design concepts and materials used in all spacecraft units will be reviewed and weight eliminated where possible.

The vehicle general arrangement is shown in Figure 3. Those items contained in Compartment D need not operate during the lunar night. There are no significant changes to the basic bus in this configuration. The resultant lateral cg location of the basic bus and its payload is as indicated in Figure 1b. This is well within the 1-in. radius adjustment capability of the retro engine.

The lighter configuration does require minor subsystem design changes:

- (1) The moment of inertia of the spacecraft has changed; therefore, a change is required to some component values in the flight control electronics subsystem.
- (2) The decrease in landed weight requires a minor modification to the landing gear shock absorber; the size of the orifice will need to be increased.
- (3) The decrease in weight of the spacecraft results in a decrease in propellant required to decelerate it; the retro-motor will be off-loaded, additional thermal liner insulation will be added in the chamber, and the nozzle shape will be altered slightly to reduce throat diameter.
- (4) The crushable landing blocks will be decreased in size.
- (5) Compartment D and its associated harness has been added to the spacecraft.
- (6) A small 5-lb silver-zinc battery has been added to the system.

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Table 4. Detail weight statement

Component	Weight, lb	Component	Weight, lb
BASIC BUS	631.7	(1) Spacecraft structure	(116.5)
Flight control system	51.6	Spacecraft primary structure	(58.5)
(1) Sensor group	(34.7)	Spaceframe	54.6
Sensor group less electronics	15.7	Landing gear pivot	1.3
Electronics	19.0	Retro adapter	2.6
(2) Secondary solar sensor	0.5	Landing gear installation	(42.1)
(3) Attitude control system	(16.4)	Landing gears 1, 2, and 3	38.5
Jet, attitude	1.6	Release mechanism	0.6
Gas supply, attitude jet	—	Crushable blocks	3.0
Uncharged	9.1	Equipment attaching hardware	15.9
Nitrogen	4.5	(2) Equipment compartments	(36.0)
Actuator, roll vernier jet	1.2	Structure assembly, Compartment A	21.4
Electronics	93.9	Structure assembly, Compartment B	14.6
(1) Data link	(30.6)	(3) Electrical wiring	(30.5)
Antenna, planar array	8.5	Wiring harness, basic bus	26.5
Antenna, omnidirectional	1.0	Thermal wrapping (Compartments A and B)	1.5
RF transfer switches	1.1	Supporting hardware	2.5
Transmitter-diplexer	12.0	(4) Pneumatic lines, attitude control	1.1
Command receiver and transponder	8.0	(5) Release mechanism, retro rocket	1.7
(2) Central command decoder	4.5	(6) Engineering measurements sensors	3.7
(3) Central signal processor	6.8	Spacecraft propulsion (not including usable propellant)	212.1
(4) Radar-altitude marker	8.3	(1) Rocket engine, main retro	138.9
(5) Doppler velocity sensor and altimeter	(22.0)	(2) Propulsion system, vernier mounted	63.7
Signal data converter	6.6	(3) Unusable vernier propellant	7.0
Klystron power supply-modulator	6.0	(4) Helium	2.5
Altimeter-velocity sensor antenna	3.7	USABLE PROPELLANT	1337.7
Velocity sensor antenna	4.5	Retro rocket propellant	1186.4
Waveguide assembly	1.2	Vernier propellant	150.7
(6) Power control system	8.6	Pyrogen propellant (igniter)	0.6
(7) TV Camera 4	5.5	SCIENTIFIC PAYLOAD	129.9
(8) Engineering data processing	6.0	TV cameras (with color)	30.6
(9) Mechanism auxiliaries	1.6	TV auxiliaries	1.3
Electrical power	53.5	Subsurface sampler and auxiliaries	39.5
(1) Solar panel	8.5	Subsurface geophysical probe and auxiliaries	4.5
(2) Battery	45.0	Temperatures (interferometer spectrometer radiometer)	2.2
Mechanisms	31.1	Scientific thermal auxiliaries	5.3
(1) Positioner, antenna and solar panel	24.6	Battery, special	5.0
(2) Antenna mechanisms, omnidirectional	4.2	Tape recorder and auxiliaries	10.6
(3) Safety and arming provisions	(1.7)	Engineering instruments	10.0
Preseparation arming device	0.5	Scientific mechanical auxiliaries	2.0
Postseparation arming device	1.2	Compartment D	4.0
(4) Separation sensing device	0.6	Wiring harness	4.0
Spacecraft vehicle	189.5	Anchors (3)	10.9

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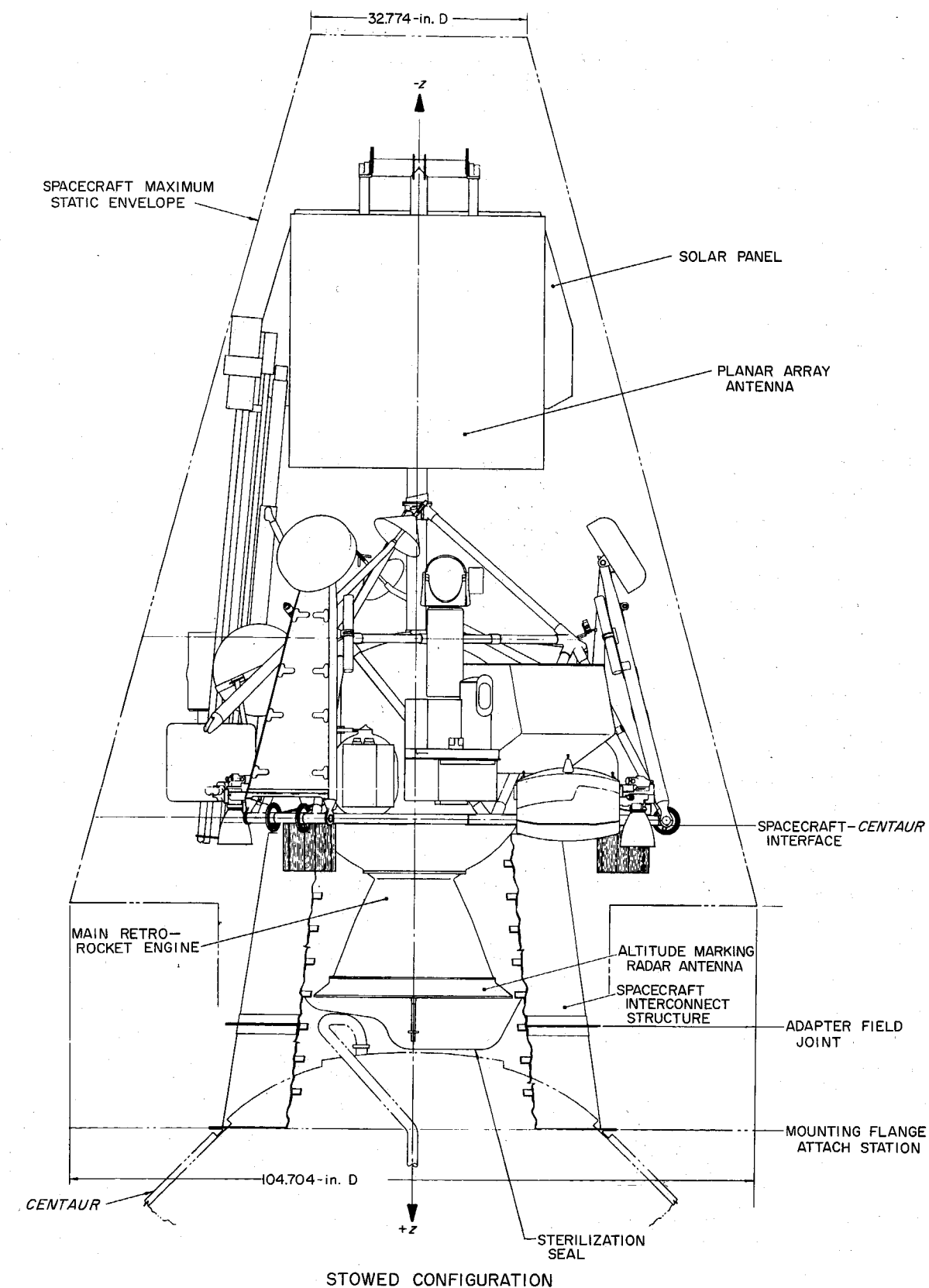
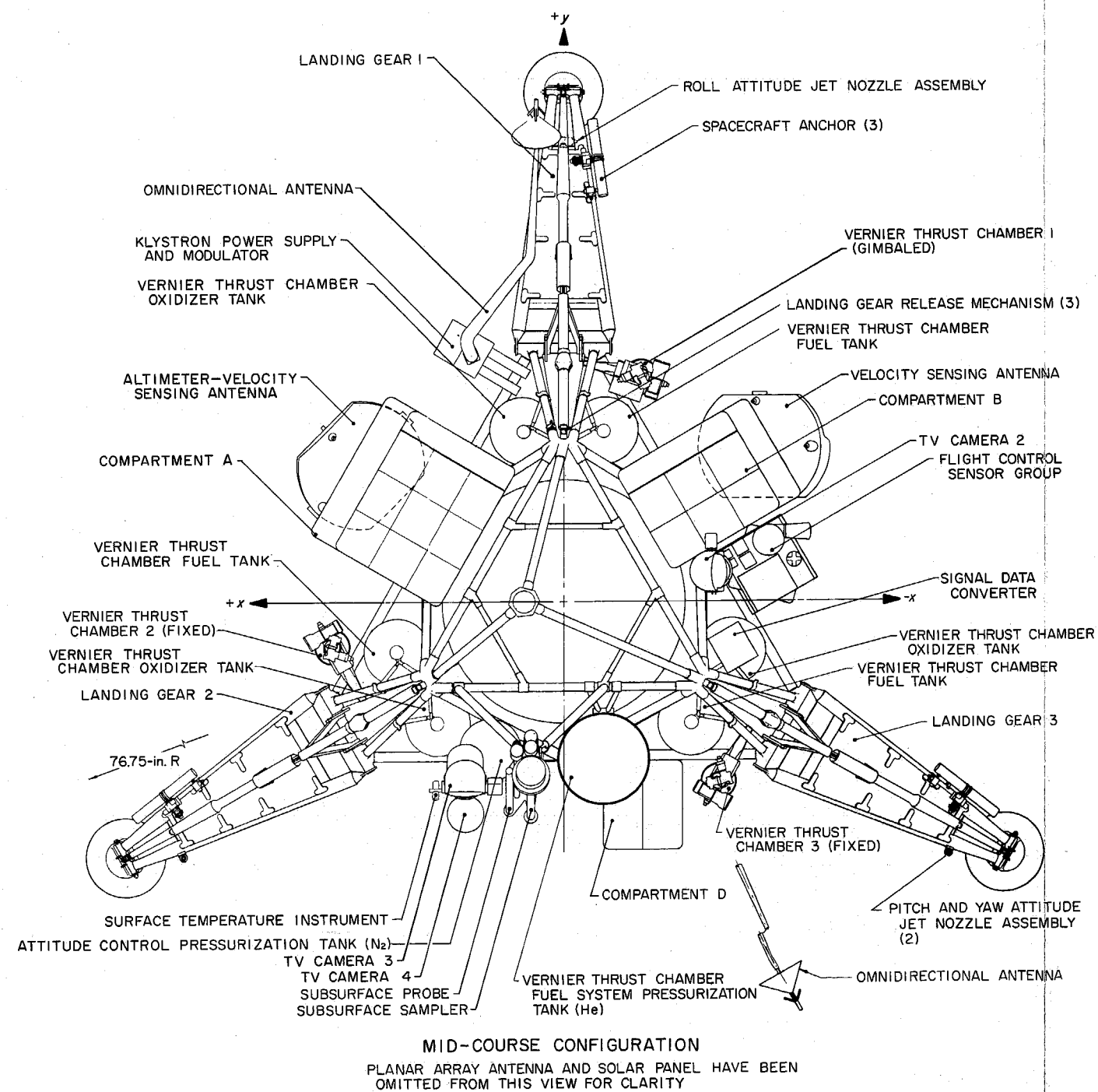
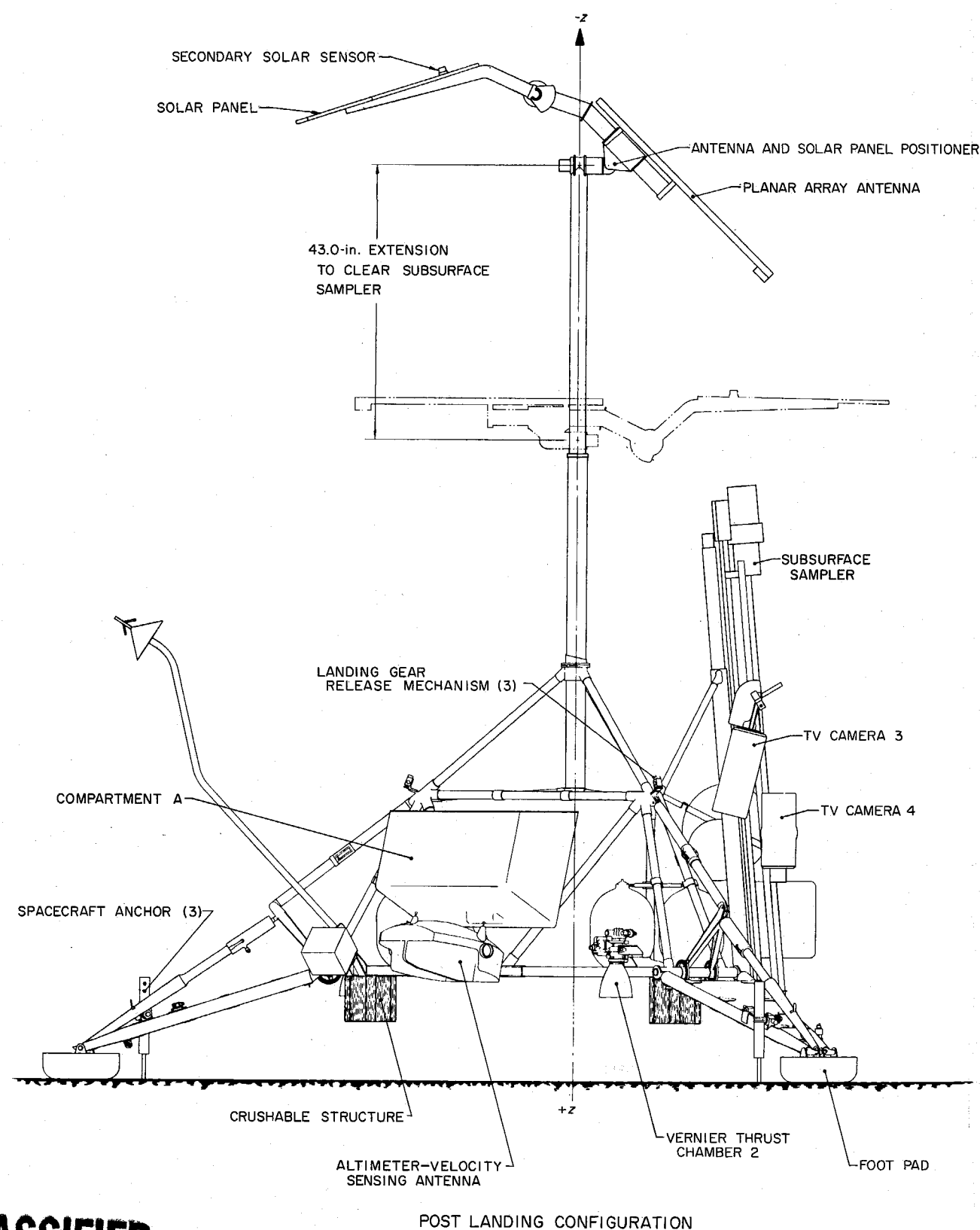


Figure 3. General arrangement of 2100-lb lightweight Surveyor

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B. Functional Performance

1. Operational Capability

The design of certain *Surveyor* modes of operation are directly dependent on injected weight, while other modes are indirectly affected depending on the actual method of implementing the change in spacecraft weight. Certain functions or operations are essentially independent. However, for the critical weight dependent modes of operation, such as main retro, vernier descent, and touchdown, design changes can be made which ensure equal

or improved performance when compared to the 2500-lb spacecraft. Table 5 summarizes the performance capability of the 2100-lb spacecraft compared to the 2500-lb configuration.

It is seen that in all areas, except lunar operations where the scientific payload and energy storage capability have been significantly reduced, performance remains unchanged or is improved by a decrease in injected weight. Although there are no inherent limitations in landing site and launch window because of reduced weight, the desire to demonstrate basic performance implies some desired restrictions as indicated in Table 5.

Table 5. Performance summary comparison of 2100- and 2500-lb spacecraft

Landing site	No change; up to 45 deg off vertical; near vertical (≈ 15 to 20 deg from normal incidence) desired to increase probability of success.
Launch window	Slightly improved; approach velocity variation allowed has been increased 75 ft/sec.
Desired lighting	High noon or earlier in order to conduct thermal testing under worst conditions.
Transit	No change.
Terminal descent	Functionally similar; main retro average thrust decrease; vernier descent changed to take advantage of increased thrust to weight.
Touchdown	No change in velocity; 15 ft/sec vertical, 5 ft/sec lateral; no change in landing stability.
Lunar	Decreased scientific data with limited payload; engineering thermal test added; single battery limits complete lunar night operation; survival at reduced temperatures will be demonstrated.

In order to increase payload for the lightweight configuration, consideration was given to restricting launch months in which favorable approach velocity conditions would prevail. A tradeoff could then be made between propellant weight and payload. Table 6 lists all the permissible launch periods (for normal incidence) from August 1963 to December 1964. Also indicated is the approximate lowest value of maximum unbraked impact speed which must be provided in order to meet the minimal postimpact Goldstone visibility requirement of 3 hr. The payload sensitivity for the 2100-lb spacecraft with respect to the maximum V_{impact} is approximately 0.26 lb/m/sec. Thus, selecting the best possible months, which would require a 1/month launch rate, results in only a 7.8-lb increase in payload. For a fixed injected weight, the increase in payload as a function of reduction in permissible launch periods is too small when traded

Table 6. Required maximum V_{impact} capability vs launch periods

Launch period		Impact period	Estimate of V_{impact} capability required, m/sec
July	30, 1963 to August 6, 1963	August 2, 1963 to August 9, 1963	2680
August	29, 1963 to September 5, 1963	September 1, 1963 to September 8, 1963	2675
September	27, 1963 to October 4, 1963	September 30, 1963 to October 7, 1963	2665
October	27, 1963 to November 3, 1963	October 30, 1963 to November 6, 1963	2655
November	25, 1963 to December 2, 1963	November 28, 1963 to December 5, 1963	2655
December	25, 1963 to January 1, 1964	December 28, 1963 to January 4, 1964	2660
January	24, 1964 to January 31, 1964	January 27, 1964 to February 3, 1964	2665
February	22, 1964 to February 29, 1964	February 25, 1964 to March 3, 1964	2670
March	22, 1964 to March 29, 1964	March 25, 1964 to April 1, 1964	2680
April	21, 1964 to April 28, 1964	April 24, 1964 to May 1, 1964	2685
May	20, 1964 to May 27, 1964	May 23, 1964 to May 30, 1964	2685
June	18, 1964 to June 25, 1964	June 21, 1964 to June 28, 1964	2685
July	18, 1964 to July 25, 1964	July 21, 1964 to July 28, 1964	2685
August	16, 1964 to August 23, 1964	August 19, 1964 to August 26, 1964	2685
September	14, 1964 to September 21, 1964	September 17, 1964 to September 24, 1964	2680
October	14, 1964 to October 21, 1964	October 17, 1964 to October 24, 1964	2675
November	13, 1964 to November 20, 1964	November 16, 1964 to November 23, 1964	2670
December	13, 1964 to December 20, 1964	December 16, 1964 to December 23, 1964	2660

against operational problems, and therefore general launch period capability was retained in the lightweight spacecraft.

2. Terminal Descent

Since the purpose of the terminal descent is the controlled removal of energy from the spacecraft relative to the lunar surface, a change in mass of the spacecraft system requires a change in total impulse requirement.

The calculation of fuel requires in general an iterative procedure, because main engine propellant can be traded for vernier by displacement of the burnout altitude and velocity. To obtain maximum landed payload capability for any injected weight, it is necessary to utilize the full thrust capability of the vernier system, and thus minimize the vernier fuel requirements. Minimum propellant requirements and maximum payload weight can then be determined as a function of injected weight as indicated in Figure 27, Section IV. A 9000-lb average thrust with an $I_s = 294$ lb-sec/lb was assumed for the main retro phase, and the vernier propellant includes the present *Surveyor* ullage factor. The curve has been derived using the predicted flight weights which would be achieved for the basic spacecraft.

Usable main retro propellant requirements, therefore, vary from 1130 to 1428 lb as the injected weight varies from 2000 to 2500 lb. Vernier fuel can be off-loaded and the weight reflected in increased payload. With a fixed minimum thrust system, a practical lower limit is implied at the point where thrust-to-touchdown weight is equal to 1 g (lunar). For a 96-lb minimum thrust setting, the 1-g limit for the conditions in Figure 27, Section IV, occurs for a minimum touchdown weight of 580 lb. (Injected weight variation is based on propellant load changes only.) For the present *Surveyor* vernier system then, the lower limit of injected weight is approximately 2000 lb.

In order to provide easy extrapolation of lightweight-spacecraft main-retro performance to the 2500-lb *Surveyor*, it appeared desirable to change the motor without changing propellant burning characteristics. This reduced the average thrust to approximately 7200 lb, and (given the lightweight-spacecraft fixed total impulse requirement of 348,878 lb-sec) made the burning time 10.3 sec longer than with the 9000-lb thrust assumed above. This in turn would introduce a vernier fuel penalty of 6.5 lb achievable at the expense of payload (since tank capacity is not a limit).

The main retro-motor proposed for the lightweight *Surveyor* spacecraft, though not of this 7200-lb-thrust,

longer-burning-time character, approximately simulates the burning time, load factors, and propellant ballistic properties at a small expense of increased vernier fuel. A lower moment control is also reflected from a general decrease in thrust level, improving confidence in the terminal descent.

Propellant incremental loading and off-loading capability will remain about constant at ± 30 lb for the 2100- or 2500-lb system. Since it now appears unlikely that an injected weight below 2050 lb will be required, a propellant weight of 1190 lb has been specified for purposes of core design. Considering the above incremental loading capability, this will give main retro performance covering an injected weight from 2050 to 2150 lb. For the proposed lightweight configuration, a usable main engine propellant weight of 1187 lb will be required.

If predicted flight weights are used and 10 lb of vernier fuel are held in reserve, the scientific payload weight for the 2100-lb injected system indicated in Figure 27, Section IV, will be reduced to 130.6 lb.

Since the *Surveyor* vernier system has continuous variable thrust control over a 3.46:1 range, it is possible to duplicate the 2500-lb descent altitude-velocity profile for the lightweight system. If no changes were made in the flight control descent parabola mechanization, the lightweight system would follow the descent curve indicated in Figure 4, which is the normal profile for the 2500-lb system. In this case, however, the average thrust commanded for a 2100-lb configuration is approximately 80% of maximum and nonoptimum vernier descent results. Fuel consumption is, of course, decreased but additional savings can be realized if the maximum thrust is utilized. Figures 5 and 6 are descent profiles for 2000- and 2100-lb injected weight systems which utilize the full thrust-to-weight capability.

The proposed 2100-lb configuration has a nominal high approach velocity start of vernier varying from 28,000 to 38,500 ft. In comparison, the 2500-lb configuration varies from 31,000 to 45,000 ft. More reliable operation of the doppler and altimeter sensors will be realized at the lower demanded altitudes of operation. In addition, it should be noted that the burnout dispersion in altitude and velocity are slightly reduced as the injected weight is decreased. This reduction is due primarily to the slightly decreased burning time and lower uncertainty of propellant burned, which results from a decrease in total propellant required. The lower burnout velocity dispersion allows a slight decrease (≈ 10 ft/sec) in the minimum approach velocity. Because sensor altitude limitations and vernier fuel restrictions are relaxed for the

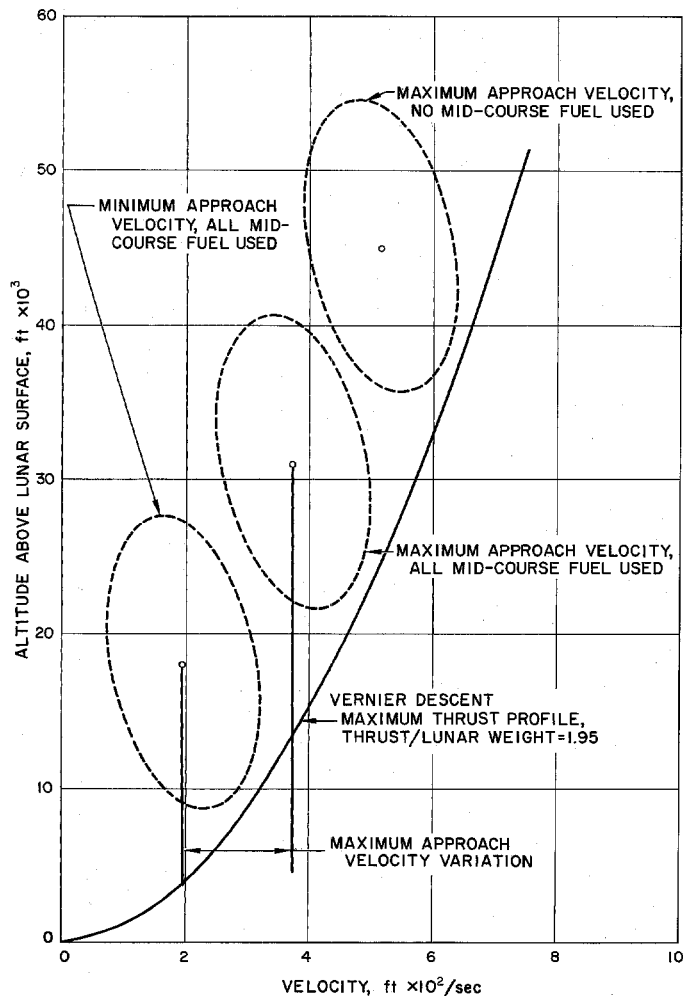


Figure 4. Vernier descent altitude-velocity profile, injected spacecraft weight = 2500 lb

lightweight descent profiles, the maximum approach velocity nominal burnout regions have been shifted 65 ft/sec higher. The above two factors have increased launch window capability by increasing the allowed approach velocity variation by 75 ft/sec. Unbraked impact speed can thus vary from 2690 m/sec (8840 ft/sec) to 2612 m/sec (8560 ft/sec). No launch restrictions, because of *Surveyor* terminal approach limitation, should exist over the full range of perigee-apogee conditions.

The increase in maximum burnout velocity is predicated on a slight change (increase) in the linear operation of the doppler velocity sensors. The present maximum range of linear operation required is 650 ft/sec. It can be seen (Fig 6) that maximum velocity at the start of vernier thrust which occurs under the condition of no mid-course fuel used will be slightly greater than 700 ft/sec. Figure 7 compares three descent profiles for the three injected weights that have been under study.

In deriving a closed loop thrust error for the vernier descent, the *Surveyor* flight control system approximates the relation $V^2 \approx 2ah$ with a number of straight line segments. Changes in the descent profile as indicated in Figure 7 require modification of the h/V slope of each segment. The slope change along with a slight decrease in the altitude at start of constant velocity and some gain changes, which may be necessary as a result of loop gain dependency on moment of inertia, appear to be the only modifications necessary in flight control to accommodate a 400-lb change in injected weight. Coast phase limit cycle behavior will also be investigated in order to ensure that adequate cold gas is available. Again damping or minor orifice changes may be necessary to accommodate the rather large change in moment of inertia.

3. Final Touchdown

The minimum thrust level of the *Surveyor* vernier engines is 30 lb per engine and was selected on the basis of a tradeoff between fuel consumption for the maximum thrust phase and ensured descent from relatively low velocity. In order to ensure adequate moment control without interaction of the thrust and attitude channels, the practical limit of total minimum thrust is 96 lb. As landed weight decreases with constant minimum thrust, the 1 g (lunar) limit is approached, beyond which it is increasingly difficult to ensure low altitude, low velocity engine cutoff and thus soft landing.

Investigation of a 2000-lb configuration, where $T_{min}/W_{min} = 1$ g (lunar), indicated that the present 5 ft/sec constant velocity phase was not satisfactory. Inability to command 96 lb or less results in a divergent descent in which cutoff altitude is not achieved. By decreasing the altitude at start of constant velocity and increasing slightly the constant velocity (7 versus 5 ft/sec), a satisfactory final touchdown can be achieved as shown in Figure 8a. In this case, the maximum vertical velocity from the engine cutoff dispersion ellipse is slightly over 15 ft/sec.

Considering the proposed 2100-lb configuration, the minimum thrust to minimum touchdown weight is 0.96 g (lunar). Maintaining the present 5 ft/sec constant velocity descent, with a ± 1.5 ft/sec dispersion, satisfactory cut-off conditions can still be achieved with a 5% variation in minimum thrust. As shown in Figure 8b, this does require a decrease of 10 ft from the present altitude at start of constant velocity.

Maximum vertical touchdown velocity is 14.6 ft/sec for both the 2500- and 2100-lb configuration. The time required to achieve stable 5 ft/sec operation will require

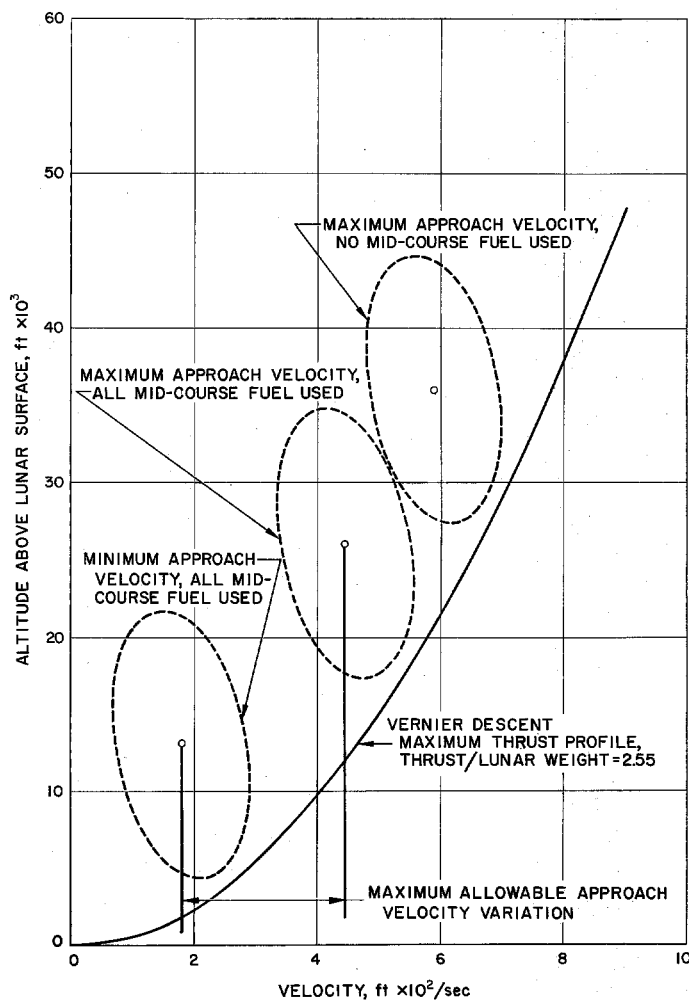


Figure 5. Vernier descent altitude-velocity profile, injected spacecraft weight = 2000 lb

some additional study and may force a slight increase over the proposed 30-ft altitude with a corresponding decrease in allowable minimum thrust variation. Since 5% appears to be more than a satisfactory margin, no change in descent velocity is anticipated.

4. Vernier Fuel

The vernier fuel required during almost all phases of engine operation is affected by the change in spacecraft weight. Only fixed time operations, such as main case separation, require a constant amount of vernier fuel. Table 7 summarizes the 2100-lb spacecraft requirements for a 48.2-sec main engine burning and a maximum thrust vernier descent as given in Figure 6. Since demonstration of soft landing is of prime importance for the proposed lightweight system, a fuel margin should be reserved to be used in any mode that might exceed its 99% condition. A reasonable reserve for such a purpose would

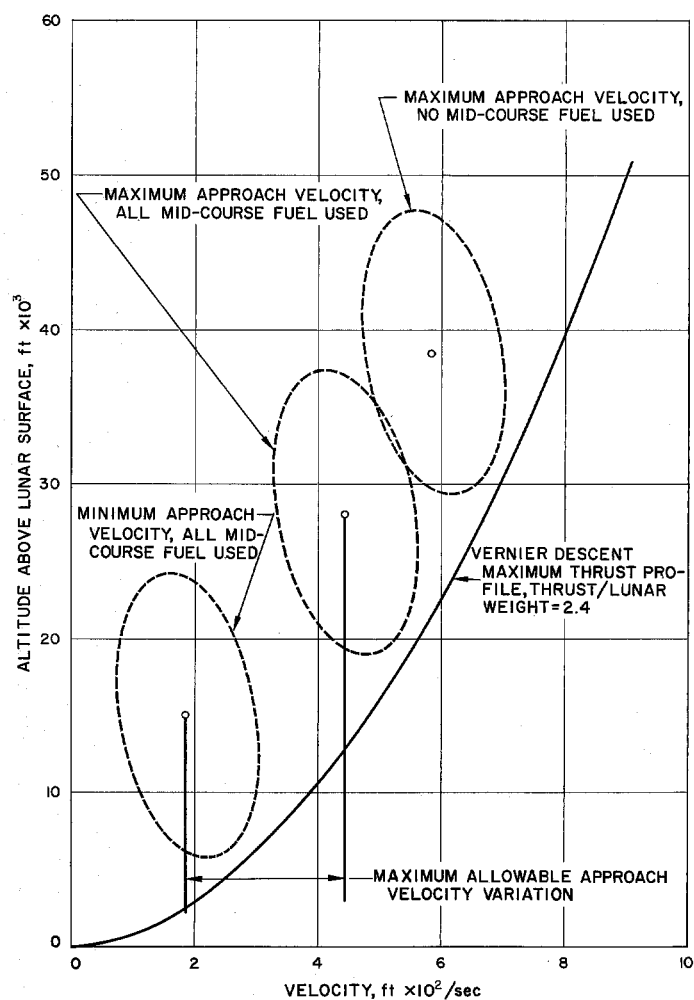


Figure 6. Vernier descent altitude-velocity profile, injected spacecraft weight = 2100 lb

appear to be 10 lb. Based on extremes of fuel consumption implied in Figure 6 and assuming 1 to 4.5 lb of nitrogen may be consumed, the minimum to maximum touchdown weights will vary from 604 to 698 lb.

5. Engineering Instrumentation

Because of the initial emphasis on a lunar soft landing demonstration associated with the lightweight configuration, a review of existing engineering instrumentation was made. Emphasis was placed on obtaining additional basic engineering data which is heavily dependent on relatively unknown environmental factors or system performance which cannot be accurately simulated on Earth.

Based on the above guideline, the terminal operations up to successful touchdown is certainly the most critical phase: sensor performance which is designed for wide variation in reflection coefficient is still predicated on a

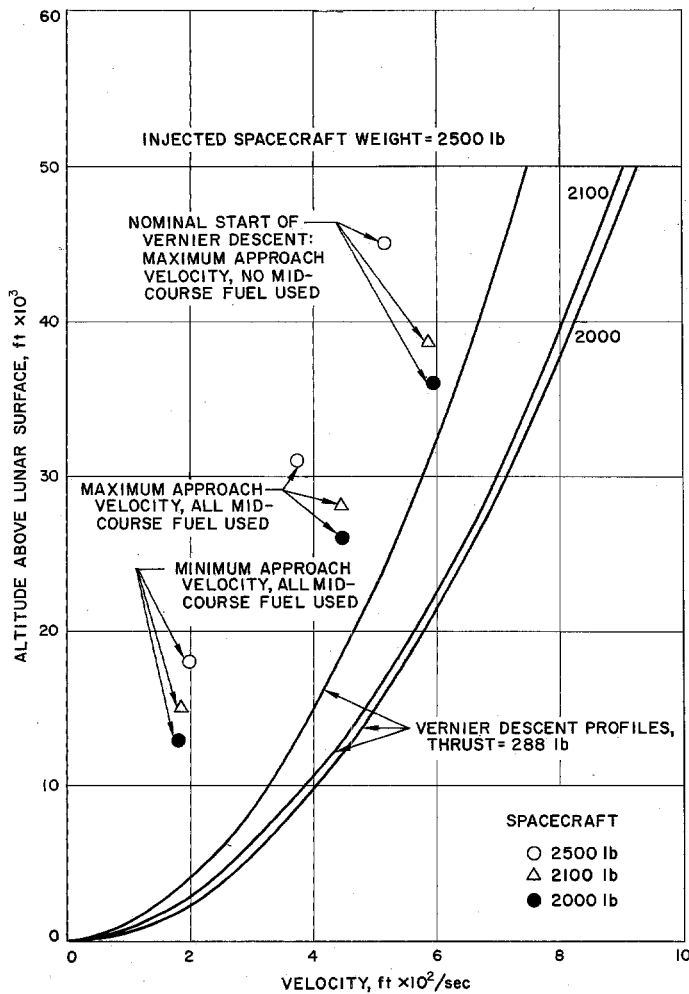


Figure 7. Vernier descent profiles for various injected spacecraft weights

relatively unknown lower limit; engine performance cannot be checked in a true vacuum; and the craft will land on an unknown surface condition with a velocity that is dependent on the total past performance of retro, vernier, and flight control. Therefore, to gain as much information as possible, complete leg and load point instrumentation is indicated. Since relatively high frequency information is desired and high gain antenna pointing cannot be guaranteed, tape recording of the information is required.

Since there are three primary load point inputs, three orthogonal accelerometers are required at each point to insure complete load definition. If, in addition, each damper is instrumented to obtain force information, 12 channels of high frequency information are required. Although load inputs with frequencies in excess of 1 kc have been specified, the calculated touchdown load is on the order of 0.1 sec with a rise time of 0.002 sec. A fre-

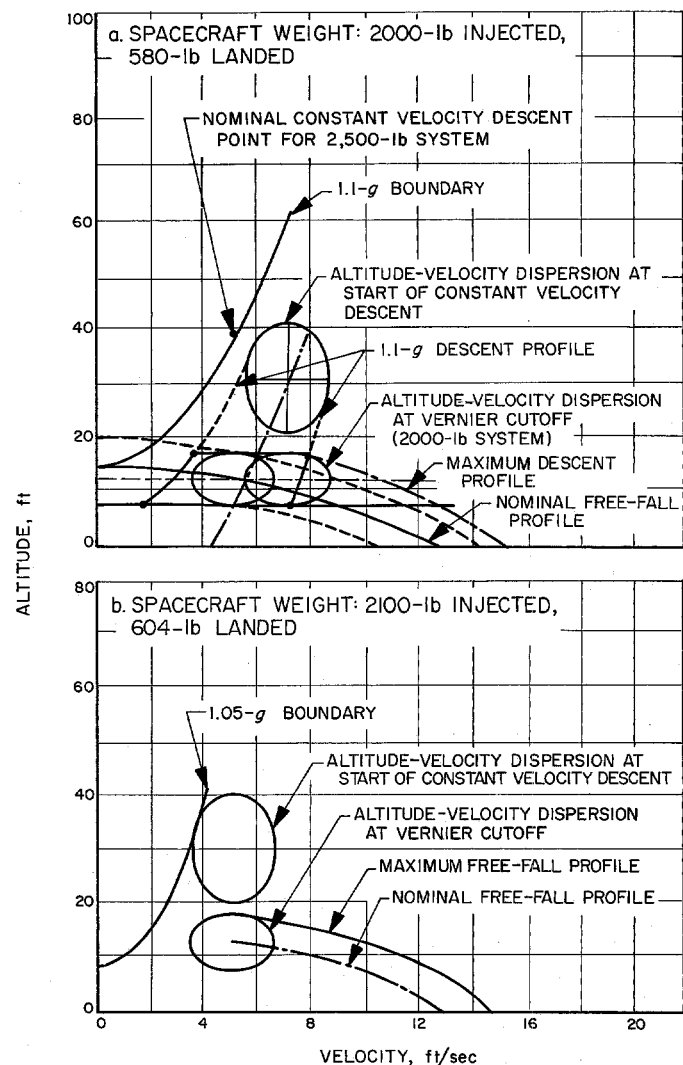


Figure 8. Touchdown velocity-altitude profiles

quency band of 300 cps should thus give adequate information on the touchdown condition. To ensure some high frequency data as backup information, three 1000 cps channels are required. In addition, the three gyro error signals are desired in order to indicate angular motion at least to the limit of the strapped down system.

The present *Surveyor* tape recorder has seven channels of 5- to 1000-cps signal bandwidth capability. In order to handle the 15 channels desired, a multiplexing system will be used. Three subcarrier oscillators and a summing amplifier will be added to four tracks so that three 5- to 300-cps signals can be recorded. Additional filters with an increased cutoff frequency will have to be supplied for the four multiplexed channels. Total weight for the above modification has been estimated at 2 lb.

Additional engineering instrumentation appears desirable in the areas of transmitter power, temperature,

Table 7. Requirements of 2100-lb spacecraft

Maneuver	ΔW (vernier fuel), lb	
	Nominal condition	Deviations
Mid-course	21.8	—
Pre-retro	0.4	—
Main retro	31	3
Case separation	7.5	1
Minimum vernier thrust	9.5	—
Maximum vernier thrust-to-touchdown (including 3 lb for descent mechanization)	61.5	8.4
Σ Nominal vernier fuel rms deviation	131.7	9.0
Reserve	10.0	
Total	150.7	

battery condition, vernier system manifold pressure, command verification, and mode conditions. A complete list of instrumentation points with additions proposed is indicated in Figure 2, Section IV. The total weight allotted for additional instrumentation will be 10 lb; therefore, some additions indicated in Figure 2, Section IV, may have to be eliminated.

6. Design Mission Sequence, Lunar Phase

The sequence of lunar operation for an eighth day launch (a launch on the last day of the monthly window) is considered for a nominal trajectory and landing location resulting in a 163-hr period of daylight before the first day-night (D-N) terminator. Under these conditions, the total available energy for spacecraft operation is 7560 w-hr. Of this, 5310 w-hr are available for scientific payload operation. The total available energy during the lunar night is estimated to be 2450 w-hr under the predicted compartment temperature profile conditions.

Immediately after touchdown, the spacecraft will be prepared for the various scientific and engineering tests to be conducted. During this period, the spacecraft attitude, functional status, and immediate environment will be determined. Extension and positioning of the planar array and solar panel will be performed. Prior to the end of the first Goldstone visibility period, the first complete TV survey and a minimum subsurface sampler operation will be completed. The initial 2-hr mapping sequence will consist of a wide angle scan with additional narrow angle scans to provide corrections to the automatic narrow angle mapping program. The initial subsurface sampler operation, which will provide a penetration of one-third of the total scheduled depth, is restricted to maintain a high state of battery charge to achieve maximum battery efficiency and life.

During the remainder of the lunar day, the basic narrow angle TV survey will be repeated periodically; each survey will be accomplished in approximately 1 hr and scheduled with increasing frequency with the approach of the terminator to take advantage of rapidly changing lighting conditions. The basic survey will consist of a 360-deg azimuth scan with each of two cameras. The elimination of spacecraft obscured areas will result in approximately 1000 narrow angle frames for each comprehensive survey. A limited amount of stereo ranging capability will be available in one sector due to duplication in the azimuth coverage. The opposite sector will be severely limited by spacecraft obstruction. In addition to the basic mapping survey, six complete color surveys will be completed, each of which will consist of 3000 frames, to provide a total of approximately 33,000 TV pictures from the mandatory mapping sequences. Up to 13 hr of discretionary TV coverage will also be available for further mapping sequences, subsurface sampler viewing or vehicle observation. It may be desirable to utilize a portion of this energy for additional experimentation, such as sampler operation and geophysical or engineering measurements. After each TV operation, the interferometer spectrometer will be directed to view the selected lunar surface area for periodic surface temperature measurements.

Subsequent to the first drilling period, the subsurface geophysical probe will be inserted in the drilled hole. If granite type of lunar material is encountered, the initial drill hole depth will be approximately 7 in. This shallow depth will limit the subsurface logging sonde data to a measurement of temperature and thermal diffusivity. The density, magnetic susceptibility, and hole diameter sensors will remain above the lunar surface. If a sandstone type of material is encountered, a penetration of approximately 20 in. will permit measurements by all the above sensors. After the second drilling, which will penetrate from 21 to 60 in. below the surface, the probe will be reinserted into the drilled hole. One complete series of measurements will then be made at selected levels of depth after which the probe will be positioned for periodic temperature sampling for the remainder of the lunar day and active portion of the lunar night.

Two engineering thermal tests will be completed for the purpose of verifying, directly and in the actual lunar environment, the thermal characteristics of the spacecraft temperature controlled compartments. Spacecraft operation during these periods are restricted to maintain, as nearly as possible, a constant dissipation of power within the compartments. The first thermal test must be conducted within 2 to 3 days of lunar noon to provide

maximum compartment solar loading. The test involves the dissipation of known power levels within the compartments and the mechanical positioning of the solar panel which must be maintained during the test. The test may be conducted under one of two possible conditions. The first could be a mast down condition for which the solar panel would be positioned by movement about the solar panel and roll axes to a point directly over the compartment. Under such conditions, several problems might be encountered. The spacecraft orientation might be such as to result in the interference of the required planar array positioning by the drill or a constraint on the operation of TV by the proximity of the solar panel to Compartment A. The mast cannot be lowered once raised, since it becomes locked in the elevated position.

The mast down condition would require minimum energy for the test but maximum temperature decay time before normal operation could resume. The second test condition could be with the mast raised. Under such a condition, maximum energy would be required for the test, although minimum temperature decay time would be needed before normal operation could resume.

The capacity of the single battery is not sufficient to provide the energy necessary to maintain the spacecraft in an active status during the entire night. To conserve energy, the thermally controlled compartments will be maintained at the minimum required temperature of 0°F for approximately 250 hr beyond the D-N terminator. During this period, the second engineering thermal test will be conducted and scientific temperatures sampled periodically. Because of the resultant lowering of compartment temperatures the spacecraft will assume a dormant status upon battery depletion. At the end of the lunar night, due to the long compartment thermal time constants, the temperature of Compartment A (which contains the battery, transmitters, and command receivers) will have dropped to only approximately -50°F while Compartment B (which contains such essential equipments as the central command decoder, central signal processor, engineering signal processor, and power control system) will have dropped to approximately -114°F. Preliminary tests indicate that these functions will survive such a temperature environment.

With the beginning of the second lunar day, the solar panel, which will have been positioned prior to the D-N terminator to receive the solar energy, will provide energy to heaters to rapidly raise the compartment temperatures. Normal compartment temperatures are expected to be achieved approximately 32 hr beyond the N-D terminator, at which time spacecraft will revert to active status and optimum battery charging will begin. All

spacecraft and scientific instrument functions will then be operated to verify night survival. TV operation and the monitoring of lunar and spacecraft temperatures will continue as operating reliability data are obtained.

The operation of the spacecraft during the lunar phase from touchdown through the beginning of the active phase of the second lunar day is summarized in Figure 9.

C. Design

As indicated previously, the lightweight spacecraft configuration requires changes in certain vehicle parameters and design changes in several of the spacecraft subsystems. Studies have been made to determine the magnitude of the changes on the vehicle hardware and their effect on vehicle performance.

1. Payload Description

The payload selected for the 2100-lb spacecraft is shown in Table 1. This payload must serve the following primary functions:

- (1) Position the spacecraft lateral center of gravity.
- (2) Provide engineering diagnostic information.
- (3) Provide maximum scientific information.

Since the position of the main retro rocket is restricted, the possible variation in the position of the thrust axis is small. Therefore, in order to keep the main retro rocket thrust alignment within allowable limits, the spacecraft lateral cg must be controlled to within approximately 0.5 in. from the spacecraft geometric centerline. With no payload, the lateral cg of the basic bus is approximately 4.5 in. from the spacecraft centerline (Fig 1). The payload selected must be so distributed that the spacecraft cg will be repositioned to within the required tolerance. To achieve this objective on the reduced weight spacecraft, a special Compartment D is added to the spacecraft in the position normally occupied by the soil analysis instruments. This compartment will house the tape recorder and its auxiliaries, subsurface sampler auxiliaries, the subsurface geophysical probe auxiliaries, scientific mechanism auxiliaries, and the special battery. None of these items are required to operate during the lunar night.

The remainder of the payload is installed in a manner identical to the installations presently being designed for the 2500-lb spacecraft. The payload arrangement is

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shown in Figure 3. With the payload so distributed, the total spacecraft center of gravity falls within the required tolerances (Fig 1b).

Engineering instrumentation is carried as part of the payload. Much of this instrumentation is intended to provide information at the time of vehicle touchdown. Because of telecommunication bandwidth limitations, all of the information desired at the instant of touchdown cannot be transmitted in real time; thus it is necessary to record it for later transmission.

The scientific portion of the payload consists of two TV cameras, the subsurface sampler, the subsurface geophysical probe, and the surface temperature experiment (interferometer spectrometer radiometer).

Compartment D. This compartment will be composed of a central structural member to which the internal units are fastened, and a light outer cover. It will be wrapped with approximately 0.5 in. of super insulation to maintain temperature limits during transit. It will be controlled by means of switches and internal heaters to control transit and daytime temperatures. The compartment temperature will not be controlled at night. The items within the compartment will, therefore, not be protected from the very low temperatures associated with the lunar night. Compartment D will be mounted near the subsurface sampler in the region normally occupied by the soil analysis instruments.

Engineering instrumentation (described previously).

Tape recorder and auxiliaries. The tape recorder will be identical to the recorder presently being developed for the 2500-lb spacecraft. It will be located in the new Compartment D. The tape recorder auxiliaries will also be installed in Compartment D.

Special battery (described in this section, under Power System).

Television cameras. Two top side cameras will be carried, both equipped with filter wheels, which may be used for color or polarimetric filters. These cameras will be mounted in the positions presently occupied by Cameras 2 and 3. The television auxiliaries will be housed with the scientific thermal auxiliaries (described below) and mounted in Compartment A exactly as installed in the 2500-lb spacecraft.

Subsurface sampler. The subsurface sampler will be the unit presently being developed to drill a 1½-in. diameter hole 5 ft deep in the lunar surface. It will be installed exactly as presently planned for the 2500-lb spacecraft.

Subsurface geophysical probe. This group of experiments will be inserted in the drill hole to make physical parameter measurements in the lunar surface. There will be no change in the installation of this unit.

Anchors. It is presently planned to carry three anchoring devices, one on each spacecraft leg in a manner identical to the present design. The anchors will be used to stabilize the spacecraft during the drilling operation, and during the insertion of the subsurface geophysical probe in the drill hole.

Interferometer spectrometer radiometer. This instrument will be mounted on the mirror of TV Camera 3 and will be used to measure the lunar surface temperature. The instrument ancillaries will be housed in Compartment A. The installation of this unit and its ancillaries will not change from the 2500-lb spacecraft design.

Scientific mechanism. This unit will be used to provide the operation logic for the subsurface sampler and will be installed in the new Compartment D.

Scientific thermal. This unit will contain a chopper stabilizer amplifier and mechanical commutator for handling thermocouple and thermopile signals from the subsurface probe. It will also contain a constant current generator and electronic commutator for resistance thermometers in the probe. In addition, it will contain any necessary heater switching for the interferometer spectrometer radiometer. This unit is housed with the TV ancillaries and is located in Compartment A exactly as installed in the 2500-lb spacecraft.

Harness. The basic bus harness will remain essentially unchanged from the 2500-lb version. The only modifications will be due to the changed payload. The payload harness will use portions of the harness being designed for the payload on the 2500-lb spacecraft. In addition, Compartment D will require a new section of payload harness to provide power, commands, and data channels for the equipment housed in this compartment.

All items will be installed in the identical location and manner as presently being designed for the 2500-lb spacecraft, with the exception of those items located in Compartment D. This arrangement, therefore, does not require extensive redesign from the present 2500-lb configuration.

A review has been made to determine if the payload could be easily off-loaded to a 2000-lb configuration. It has been determined that this cannot be easily done. In order to properly balance the vehicle lateral center of gravity, a major rearrangement of payload and basic bus units would be required.

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2. Thermal Control System

a. Transit and lunar day. The present thermal designs for Compartments A and B will assure 0 to 125°F thermal control for the equipment housed in Compartments A and B in the 2100-lb configuration. Compartment D, a new compartment, will have two thermal switches set at a nominal 70°F contact closure temperature. Due to the absence of operating equipment (electrical power dissipation) in Compartment D during transit, approximately 2 to 3 w of auxiliary heating will be required to maintain equipment bulk temperatures above 0°F at the end of transit (Fig 10). During lunar operation, the lunar environmental input, in conjunction with normal power dissipation generated from the subsurface experiments, will be sufficient to maintain Compartment D equipment temperatures in the range 0 to 125°F.

b. Lunar night.

Compartments A and B. Assuming that 2450 w-hr of battery energy are available for lunar night compartment thermal control, a lunar night mission sequence, making maximum use of the compartment transient response, has been established. This sequence will yield a maximum output of engineering data on the compartment thermal control system. Since engineering data on compartment thermal performance should be obtained at 0°F (which is the lower limit of acceptable equipment performance), it is desirable to maximize the time that the compartments can be maintained at 0°F with the available energy. Figure 11 shows the cooling rates of the compartment equipments from D-N terminator. Since the response of the individual compartments varies, Compartment B will

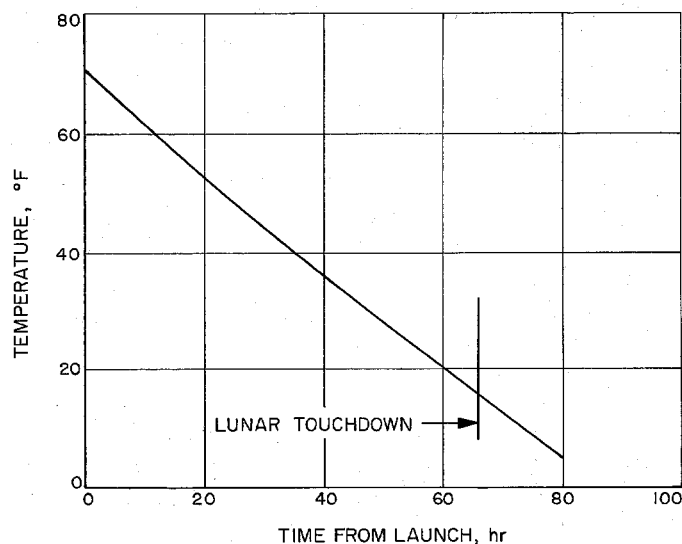


Figure 10. Compartment D bulk equipment temperature

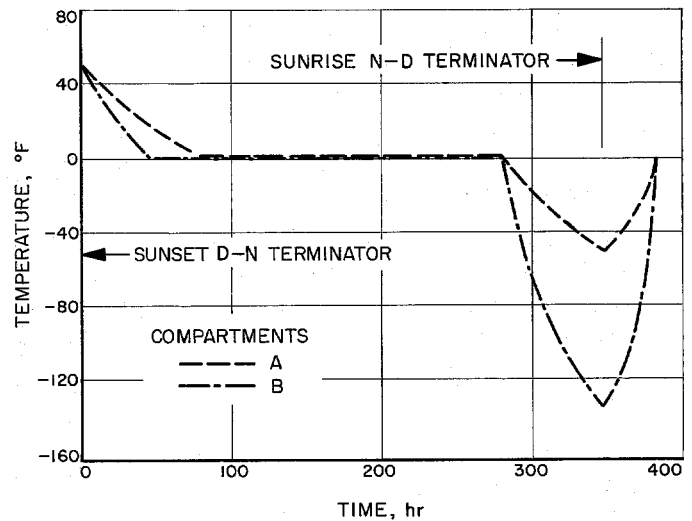


Figure 11. Compartment temperature during lunar night

reach 0°F prior to Compartment A. The minimum amount of power required to maintain an RF link should be dissipated during the transient period prior to reaching 0°F compartment equilibrium temperatures. The compartments will contain auxiliary heaters which can automatically control each compartment to 0°F with 5 w of power. Based on the 2450 w-hr of battery capability, a minimum of 200 hr of 0°F temperature control is assured during which engineering data for evaluation of the thermal control performance can be gathered. Since the available energy is inadequate to provide complete lunar night protection for both compartments during the 70 hr remaining until sunrise, Compartment A and B equipment will cool to -50 and -115°F, respectively.

With the exception of the transmitters and command receivers, all basic bus electronics are being designed to withstand -300°F survival temperatures. Initial test results have indicated that the transmitters and command receivers can readily survive -50°F, and all basic bus electronics will likely be usable the second day.

At sunrise, electrical power from the solar panel will be available to reheat Compartment A and B equipments. The logic of the power control and heater systems will bring the heaters into use as soon as the solar panel is energized.

Compartment A equipment will reach 0°F at 32 hr and, similarly, Compartment B equipment will reach 0°F 28 hr after sunrise. Normal operation of the basic bus electronics in Compartments A and B is possible 32 hr after sunrise.

c. Compartment D. This compartment will not be thermally controlled during the lunar night, and therefore

will experience temperatures in the vicinity of -300°F . Depending upon the survival capability of the enclosed equipment, the compartment may have second day operational capability.

3. Touchdown Dynamics

Preliminary calculations have been completed showing the feasibility of landing a lightweight vehicle while keeping loads within the existing limit levels and retaining acceptable touchdown stability characteristics. So far all calculations have been made considering a 580-lb touchdown weight as compared to the nominal weight of 772 lb for a spacecraft of 2500-lb separated weight.

In order to maintain landing loads at acceptable levels it is necessary to reduce the compression stroke damping of the shock absorber and the crushing force of the crushable body blocks. The zero deflection damping constant has been reduced from $0.48 \text{ lb}/(\text{in.}/\text{sec})^2$ to $0.324 \text{ lb}/(\text{in.}/\text{sec})^2$. The crushing force of each block has been reduced from 2700 to 2070 lb.

Calculation of maximum antenna lateral accelerations and package accelerations have been completed; package acceleration refers to the vertical acceleration of a point displaced laterally 30 in. from the cg.

Antenna lateral accelerations were made using the following values:

Radius of gyration	28 in.
Δcg	18 in. (height of vehicle center of gravity above the landing gear hinge line, used as reference plane)
Friction coefficient	1.0
Site slope	15 deg
Amplification factor	2

The results using a touchdown vertical velocity of 20 ft/sec are shown in Figure 12. On the basis of a 3σ flight control dispersion ellipse the maximum acceleration level is approximately 22 g. The same value also results from considerations of a 10σ ellipse with touchdown vertical velocity of 15 ft/sec (Fig 12).

Maximum ft/sec package accelerations were run using a friction coefficient of zero and a $\Delta \text{cg} = 16.05 \text{ in.}$ Parameters otherwise were as quoted above. Results for a 20 ft/sec touchdown vertical velocity are shown in Figure 13. On the basis of a 3σ dispersion ellipse a maximum acceleration of 25 g is established. Thus, the acceleration

levels for the lightweight vehicle are lower than those for the standard (772 lb) vehicle.

The reduction in damping results in an automatic reduction in shock absorber loads. The input loads to the spaceframe are therefore lower than the loads being used for design of the 2500-lb system.

The relatively stronger spring results in slightly better stability characteristics as shown in Figure 14 which uses a $\Delta \text{cg} = 18.0 \text{ in.}$ Although only this one stability boundary has been investigated, this comparison should be representative of the results that will be found for all variations in friction, center-of-gravity, site slope, etc. It should be noted that the increased stability of the 580-lb vehicle could permit increases in vertical cg height of about 2 in. relative to the 772-lb vehicle, or alternatively a 2-deg increase in site slope capability.

Final weight and balance calculations have indicated that for the configuration selected, the landed weight should range between 607 and 701 lb. In view of these weights, the modification to the landing system design should be on the basis of approximately a 650-lb vehicle. The addition of the subsurface sampler in the final configuration resulted in raising the vehicle center of gravity by about 1 in. above that for the present 2500-lb configuration. This change will result in lower inertia loads on the antenna and a decrease in vehicle stability at touchdown. However, the pitch radius of gyration was reduced by 5 in. from the heavyweight vehicle, which will result in improved stability and slight increases in antenna and package inertia loads. The net result should yield design loads of the same magnitude as presently being used for the 2500-lb configuration. Thus, from a load and stability viewpoint the modified vehicle is acceptable.

The required parameter modifications can be effected with little difficulty. To reduce the block crushing force two outer rings of cells will be eliminated resulting in 313 instead of 409 cells. This also changes the length/diameter from 0.875 to 1.0, the previously stated maximum value.

The shock absorber compression stroke damping is accomplished by means of two orifices: (1) a drilled hole through the piston which gives the constant portion of the damping, and (2) the slots in the metering pin which gives the increase in damping with stroke. The required damping can be achieved by enlarging the orifice in the piston.

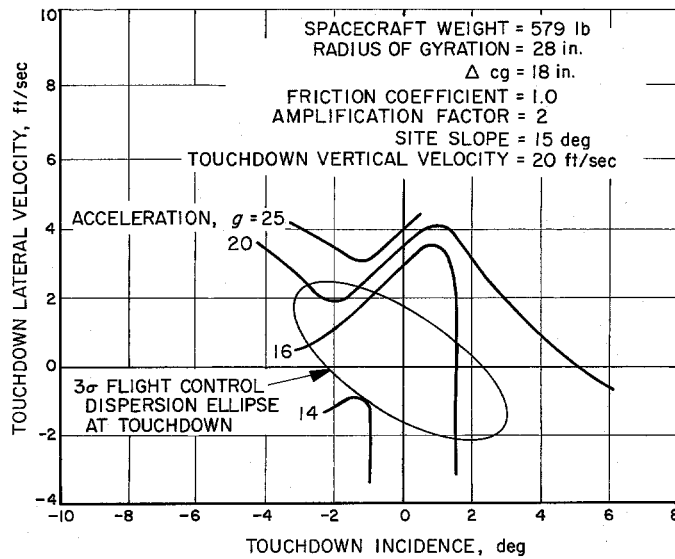


Figure 12. Antenna lateral acceleration

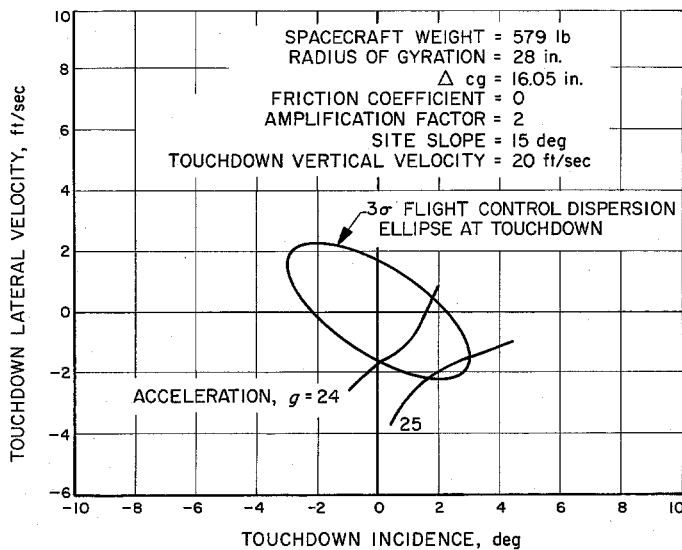
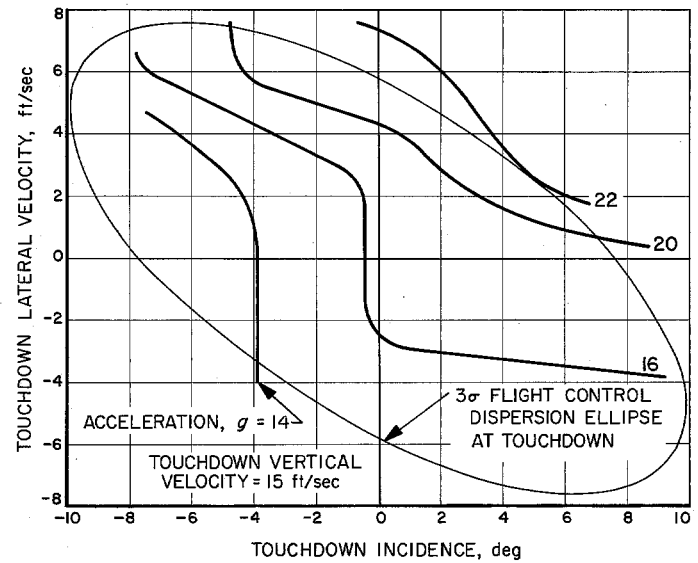


Figure 13. Package vertical acceleration

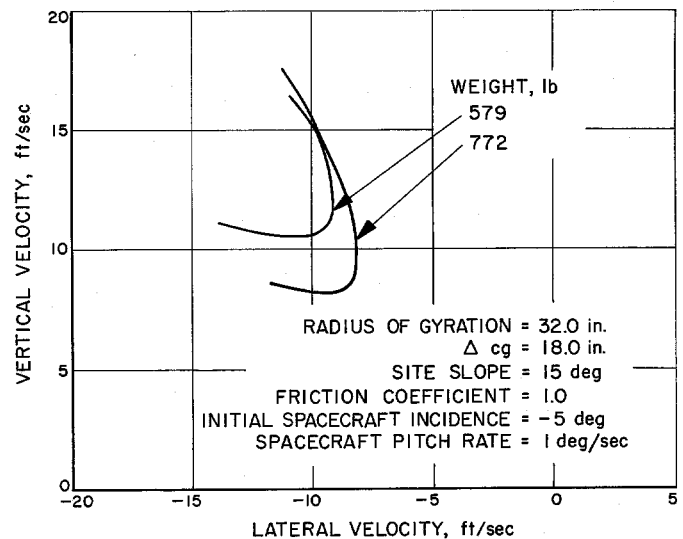


Figure 14. Effect of weight on stability boundary

4. Power System

The power system of the current *Surveyor* design consists of a solar panel for primary power, two silver-zinc rechargeable storage batteries of 2650 w-hr capacity each, and the necessary power management and regulation electronics. Since only one of these batteries is nominally required to conduct basic bus operation, the other is removed in stripping down to a minimum weight spacecraft system. The electrical power system for the light-weight design is proposed to include the following: (1) the same solar panel as in the existing 2500-lb configuration, (2) one existing sealed, secondary silver-zinc battery

weighing 45 lb with a capacity of 2650 w-hr, (3) the same power management logic and components, and (4) a 5-lb sealed secondary type silver-zinc battery to handle the retro ignition and retro separation squib-firing power requirements.

A study of peak loads demanded of a single battery revealed that the greatest requirement occurs during the terminal descent phase. When the doppler velocity sensing radar and altimeter are activated, a load of 438 w is added to the existing average drain of 300 w, a load established primarily by the high-power transmitter and flight control system. Although this steady-state load is

somewhat beyond the nominal power capacity of the single battery, its short duration of 3 min places the energy requirement well within the battery's capability.

A problem does arise from pulse requirements associated with main retro ignition and separation. Although these events are separated in time and are of relatively short duration (50 msec), redundancy in the firing circuits causes an extremely high current which exceeds the capacity of a single battery.

Partial alleviation of this problem has been accomplished by sequential firing of redundant squib banks on the retro separation system. This change is also being implemented into the 2500-lb spacecraft.

Figure 2 represents the terminal descent power requirements after the introduction of this change. A peak current of 65 to 91 amp is required for a 10-msec interval during retro separation. Although the battery supplier has indicated that 70 amp may be removed from a single battery for a period of 1 sec at a battery temperature of 70°F, it is apparent that additional measures must be taken in the lightweight spacecraft. Accordingly, it is proposed that an auxiliary battery of the sealed primary type be added to the lightweight *Surveyor* spacecraft.

Since the lunar phase of the lightweight *Surveyor* mission involves a worst case peak load condition of 65 amp for 300 msec during surge current start of the lunar drill, the added battery is not required after touchdown, and may be a high rate, low capacity device. A weight of 5 lb is realizable under these conditions, and includes the redundancy of two separate parallel cell banks in the auxiliary battery.

The battery tentatively selected to satisfy the squib ignition requirements is a sealed, silver-zinc, secondary type battery for primary use only. The battery will utilize flat-plates and thin separators to provide the capability for high current discharge rates. Weight is estimated at 5 lb and volume at 40 in.³. Approximate over-all dimensions are 4 × 4 × 2½ in. The battery will consist of two parallel-connected series strings of eight cells each. Each series string will be capable of providing at least two 50-msec pulses of 60 amp at a temperature within the range of 70 to 90°F. Voltage limits for the battery will be 9.4 ± 1.6 v.

Design information for this proposed battery has been derived on the basis of similar batteries previously designed for squib firing purposes by Electric Storage Battery Company. Availability of the proposed 5-lb battery will require only a limited amount of design and

development effort and a qualification program will be conducted to assure its performance ability.

Battery manufacturers have been consulted as to the availability and suitability of the battery required for this application, and both Eagle-Picher and Electric Storage Battery Company have proposed batteries. Electric Storage Battery proposes the use of a sealed, secondary type of silver-zinc primary battery with a high discharge rate capability, which appears at this time to be the most satisfactory.

In the present retro separation system (without the introduction of sequential firing of redundant squibs) a firing signal from the flight control programmer triggers silicon-control rectifiers which fire both banks of explosive bolt squibs. Pyrotechnic switches are also fired by the rectifiers and remove all circuits from the batteries after a suitable delay. Sequencing of the redundant squib banks can be accomplished by additional sets of normally open contacts on the redundant pyro switches in the first bank, and utilizing these contacts to fire the silicon controlled rectifier in the second bank.

In the lightweight *Surveyor* design, the auxiliary high rate battery could be inserted directly in place of the battery removed, but changes are required in charging logic and a means of removing it from the line after use. Doing so eliminates the danger of unbalance with the prime battery since the batteries are of different capacities. Investigation has also disclosed that considerably more current can be obtained in the same gross weight if the auxiliary battery is split into two 9.4-v cell banks instead of one 22-v string as in the prime battery.

Since the squib firing circuits can be easily operated at this lower voltage, it is proposed to isolate all main retro ignition and separation squibs on the auxiliary battery. In this way all possible interactions with the prime battery are eliminated and the separation circuits are redundant since either cell bank of the auxiliary battery is capable of firing peak squib loads. Provision will be made for charging of the battery immediately prior to launch. Figure 15 illustrates the proposed arrangement.

Additional studies are in progress to reduce peak squib currents by replacing silicon rectifiers with power transistors, and thus reducing the pyrotechnic switch requirements and currents. Should this prove possible, the auxiliary battery might be eliminated at a later date.

5. Main Retro-Rocket Engine

An investigation has been conducted to determine the requirements for the main retro-rocket engine having

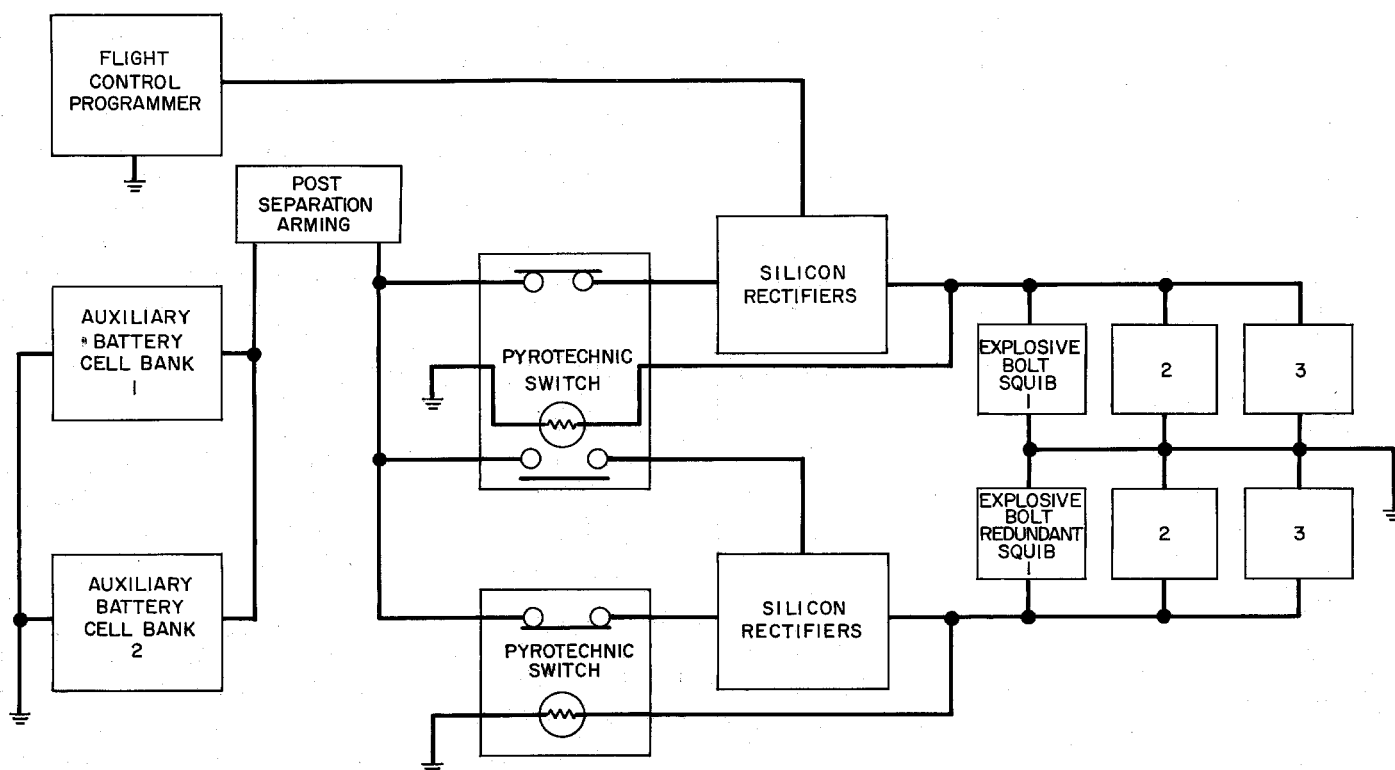


Figure 15. Proposed design of main retro separation system for 2100-lb spacecraft

the total impulse needed for the lightweight *Surveyor* spacecraft by means of minimal modifications to the current 2500-lb spacecraft engine. The following basic ground rules are:

- (1) Engines for the lightweight spacecraft will use the same propellant and igniter and the same inert components as the current engine.
- (2) Minimum reduction in average thrust from that of the current design but commensurate with the design constraint of the current engine case.
- (3) Maintain the same schedule for flight deliveries.

A summary of the weight and performance features of the modified engine for the 2100-lb spacecraft and comparison with the current engine is shown in Table 8. The corresponding thrust-time curve is shown in Figure 16. The engine will have a loading capability of ± 30 lb about this nominal value shown.

To achieve the desired ballistics, a grain redesign was required. A straight, pulled core will be used. The core diameter will be increased to a maximum consistent with the diameter of the aft port of the case. The grain cross-section is shown in Figure 17. For comparison, the grain cross-section of the current *Surveyor* main retro-rocket is

Table 8. Engine characteristics

Parameter	Engine for 2100-lb spacecraft ^a		Engine for current spacecraft
Temperature, °F	50	70	70
Thrust (max), lb	9,345	9,500	10,000
Thrust (avg), lb	8,170	8,370	9,260
Pressure (max), psi	555	569	560
Pressure (avg), psi	485	500	520
Total impulse, lb-sec	350,000	350,000	420,100
Action time, sec	43	42	45.5
Propellant specific impulse, lb-sec/lb	294	294	—
Propellant weight, lb	1190	—	—
Inert components (max), lb	141	—	—

^aAll values are for operation in vacuum; average expansion ratio = 59.5, and assumed $C_d = 0.96$.

shown in Figure 18. A reduced burning area will accompany the grain redesign, thereby requiring a smaller throat diameter to maintain the same average pressure as the present version.

Grain design investigations for the lightweight version began early in December 1961. Preliminary work was concentrated on the feasibility of obtaining propellant

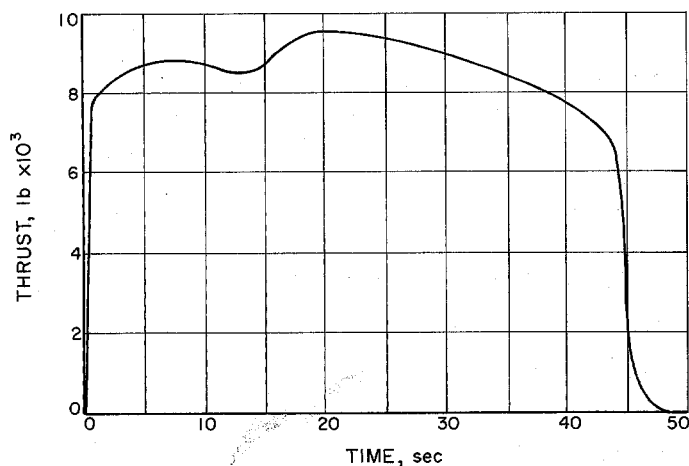


Figure 16. Thrust vs time relation for modified engine

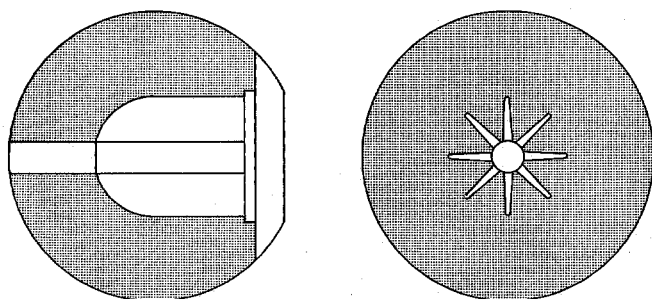


Figure 17. Main retro-engine for 2500-lb spacecraft

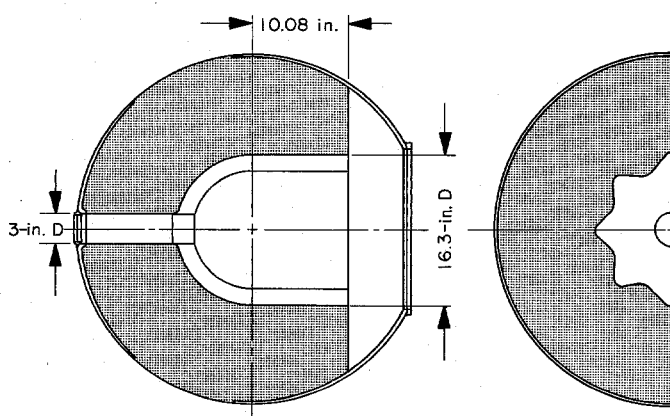


Figure 18. Main retro-engine for lightweight spacecraft

weights in the 1100- to 1300-lb range using the basic main retro engine hardware. By varying the cavity diameter and the propellant level at the aft end, the range of propellant weights obtainable could vary from 1000 lb to an optimum of approximately 1500 lb. Work was then continued to obtain a grain design with nominal propellant weight of 1190 lb which would yield thrust-time and pressure-time profiles consistent with retro-engine

performance requirements. The following additional ground rules were used:

- (1) The maximum thrust requirement of 10,000 lb shall be applicable to a 70°F bulk grain temperature.
- (2) The desired average thrust shall be 9000 lb.

Two grain design concepts predicated on the above guidelines were investigated. The first concept was based on a design which was defined by allowing the 1428-lb version to burn 240 lb of propellant starting from the initial burning surface. The resulting grain design would then be identical to a specific grain surface in the 1428-lb version minus 240 lb. Direct correlation of burning curves would be possible starting at some discrete time on the 1428-lb burning curve. The primary consideration in this design was the requirement for a collapsible or meltable core due to engine case aft flange diameter limitations. The second concept involved a similar grain design as the 1428-lb version with the desired propellant weight being obtained by increasing the cavity diameter to the maximum allowed by the flange diameter and by reducing propellant level of the aft end of the engine. The primary consideration in this design was the difference in burning curves which would result between the 1428- and 1190-lb versions. The second concept was chosen based on the following considerations:

- (1) Meltable and collapsible cores require the development of suitable production techniques from essentially developmental concepts.
- (2) The difficulty of maintaining dimensional control with meltable and collapsible cores.
- (3) The production cycle for incorporation of a straight-through core into the present program plan would remain the same. Additional cost and production cycle time would be required by meltable and collapsible cores.

The first cut at a straight, pulled core grain design yielded a circular port-head end web design as depicted in Figure 19; the corresponding performance characteristics are shown in Figure 20. In consideration of present propellant casting techniques, the design was modified to include star points to facilitate insertion of the casting tubes.

The final 1190-lb grain design is presented in Figure 17. The 9000-lb average thrust requirement was not met due to limitations on the maximum thrust of the 1428-lb version.

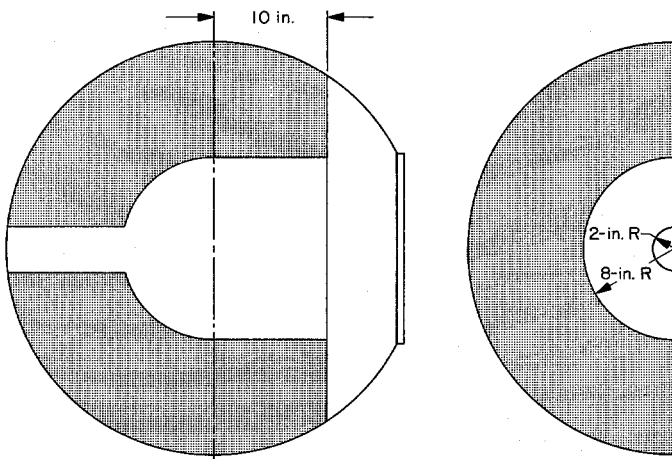


Figure 19. Circular port, head-end web design

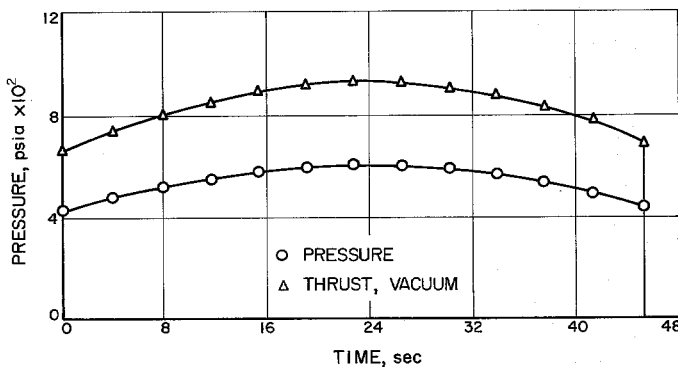


Figure 20. Performance characteristics of circular port, head-end web design

D. Reliability

The expected reliabilities of the proposed lightweight *Surveyor* spacecraft have been estimated for both the flight and landing and first 80 hr of lunar operation. Using the same methodology as presently developed for the standard spacecraft, and the latest available failure rate data pertaining to proposed parts, it is estimated that the system reliability (i.e., the probability of successfully performing the planned essential hardware functions) will be at least 0.67 (80% confidence) in the flight and landing phase and at least 0.90 (80% confidence) during the first 80 hr of lunar operation. The reliabilities of individual experiments are not included in these figures.

The equipment group contributing most to the unreliability of the proposed lightweight spacecraft is the electrical power supply which comprises one 45-lb, plus one 5-lb battery, for the squib circuits. The standard spacecraft presently uses two 45-lb batteries in parallel. Potential unreliability in this area can be reduced by an evaluation test program.

A comparison of equipment groups of the lightweight spacecraft with the 2500-lb spacecraft is given in the following:

Flight control system (FCS). The FCS is essentially identical with the present system intended for use in the standard spacecraft. Considerable effort has already been expended in obtaining estimates of this system. For this reason the values used for the lightweight spacecraft are the same as those already reported for the 2500-lb spacecraft. The FCS is not needed during the lunar phase.

Propulsion system. The proposed vernier engine is the same as the one used in the present standard spacecraft. The proposed retro-engine is basically similar except for the fuel quantity needed to accomplish the mission of the proposed spacecraft. It is felt this will not affect reliability, and therefore the quantized values are the same for the lightweight as for the standard spacecraft. The propulsion system is not needed during the lunar phase.

Electric power. A single 45-lb battery is proposed for the lightweight spacecraft supplemented by an additional 5-lb battery to handle peak loads during lunar landing. The predicted mean reliability of a single 45-lb battery is 0.989 with the 80% confidence limit at 0.038. Since the supplemental 5-lb battery will be an internally redundant configuration, degradation in the spacecraft failure rate during transit and descent phase will not be significant. It is presently felt that normal power requirements for the lunar phase can be satisfied with the 45-lb battery alone.

Electronics. It is proposed to use essentially the same electronics system as in the present standard spacecraft; it should be pointed out that such reliability estimates do not include the electronics for scientific experiments.

Vehicle. The slight degradation in the vehicle reliability values for the lightweight spacecraft is due to the additional wiring for connecting to the added compartment. Present harness for essential equipment weighs approximately 30 lb. Harness for the proposed lightweight spacecraft will weigh about 34 lb. It is assumed the failure rate increases in about the same proportion.

Mechanisms. Both the standard vehicle and the lightweight vehicle reliabilities include the weighted average

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of significant manipulation equipment used in experiments (such as the drill motor), solar panel drives and the anchors, and for the lightweight design are essentially unchanged from the 2500-lb system.

Thermal control. It is assumed the thermal group failure rate could be increased by a factor proportional to the additional compartment in the proposed lightweight spacecraft. Because the resulting reliabilities are very close to unity, the effects of the degradations are negligible in the calculation.

Systems. The mean system reliability figures are obtained essentially by means of the product rule. The system reliability limits were obtained by root-sum-square procedures presently followed in estimating *Surveyor* reliabilities. Estimated reliabilities show a negligible effect due to the configuration change of the battery and harness during flight and landing. The battery degradation is the most significant item during the lunar phase; however, the resulting mean reliability value still assures better than 95% probability of successful hardware operation.

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PART THREE

PLANETARY-INTERPLANETARY PROGRAM

I. Objectives

The primary long range objective of the NASA Planetary-Interplanetary Program is the development of automatic, unmanned, interplanetary spacecraft technology and the use of this technology in the form of space probes to gather fundamental scientific knowledge concerning the planetary and interplanetary environments, the planets themselves, and solar phenomena, both out of and within the plane of the ecliptic.

The secondary long-range objective of the program is the development of technology and the collection of scientific data which will contribute to the successful manned exploration of the planets and interplanetary space.

The primary objective of the program by 1970 is to have rather completely demonstrated and, to some reasonable extent, exploited (in terms of acquired scientific

data) spacecraft capable of (1) being put into orbit around and (2) landing on the surface of Mars and Venus.

A secondary objective during this period will be the initial effort toward extending the above capability toward the planets Mercury and Jupiter and space shots out of the plane of the ecliptic and in toward the Sun. The immediate objective of the program is the initial probing of Mars and Venus.

The Planetary-Interplanetary Program at present consists of three projects: *Mariner R* using the *Atlas-Agena B* vehicle, *Mariner B* using the *Atlas-Centaur* vehicle, and the *Voyager*, using the *Saturn* vehicle. The missions planned for these projects are obviously subject to change because of such influences as new scientific discoveries or unexpected developments in vehicle availabilities.

II. Mariner R Project

The intent of the *Mariner R* project is to perform flyby missions to Venus in 1962. These missions are a replacement for the *Mariner A* project which was cancelled during September 1961 due to unavailability of *Centaur* launch vehicle.

The primary objective of the *Mariner R* project is to develop and launch two spacecraft to the near vicinity of the planet Venus, to receive communications from the spacecraft while in the vicinity of Venus, and to perform a radiometric temperature measurement of the planet. A

people to discuss and review current status of each system, and to bring to light any remaining problem areas.

The *Project Development Plan*, reviewed and agreed upon by Marshall Space Flight Center and JPL, has been submitted to NASA Headquarters for final approval.

A. Systems Test and Operation

The *Mariner* MR-1 and -2 assembly and operations have essentially proceeded according to schedule (Table 1). Early delivery of the mechanical equipment allowed the

Table 1. Spacecraft assembly and operations schedule

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start of assembly 1 week early. Punctual delivery of most other equipments contributed to near completion of the assembly phase in 3 instead of the allotted 4 weeks. The time picked up was then effectively used to resolve the problems encountered in the subsystem and interface test.

The MR-1 and -2 test complex is shown in Figure 1.

1. Mariner MR-1

a. Mechanical assembly. The mechanical assembly of *Mariner* MR-1 (Fig 2) was started early in January 1962. Completed by late January, assembly status was sufficient to start electrical checkout of the spacecraft.

b. Electronic and electrical checkout. The first phase of spacecraft electrical checkout is the verification of wiring and grounding integrity. This was accomplished

with little difficulty, although several ground loops were discovered and corrected.

The second phase of the spacecraft electrical checkouts begins with initial power turn-on. Although this phase is critical, no troubles occurred during application of power to the spacecraft.

The third phase of checkouts involves the testing of all subsystems.

The fourth phase of *Mariner* R Spacecraft Assembly Facility (SAF) testing is the interface checks. An interface is defined as the mating of signals between subsystems.

The fifth phase of SAF testing is telemetry calibrations. The telemetry calibrations have been conducted with essentially no problems. All calibrations have been completed with the exception of the antenna reference hinge

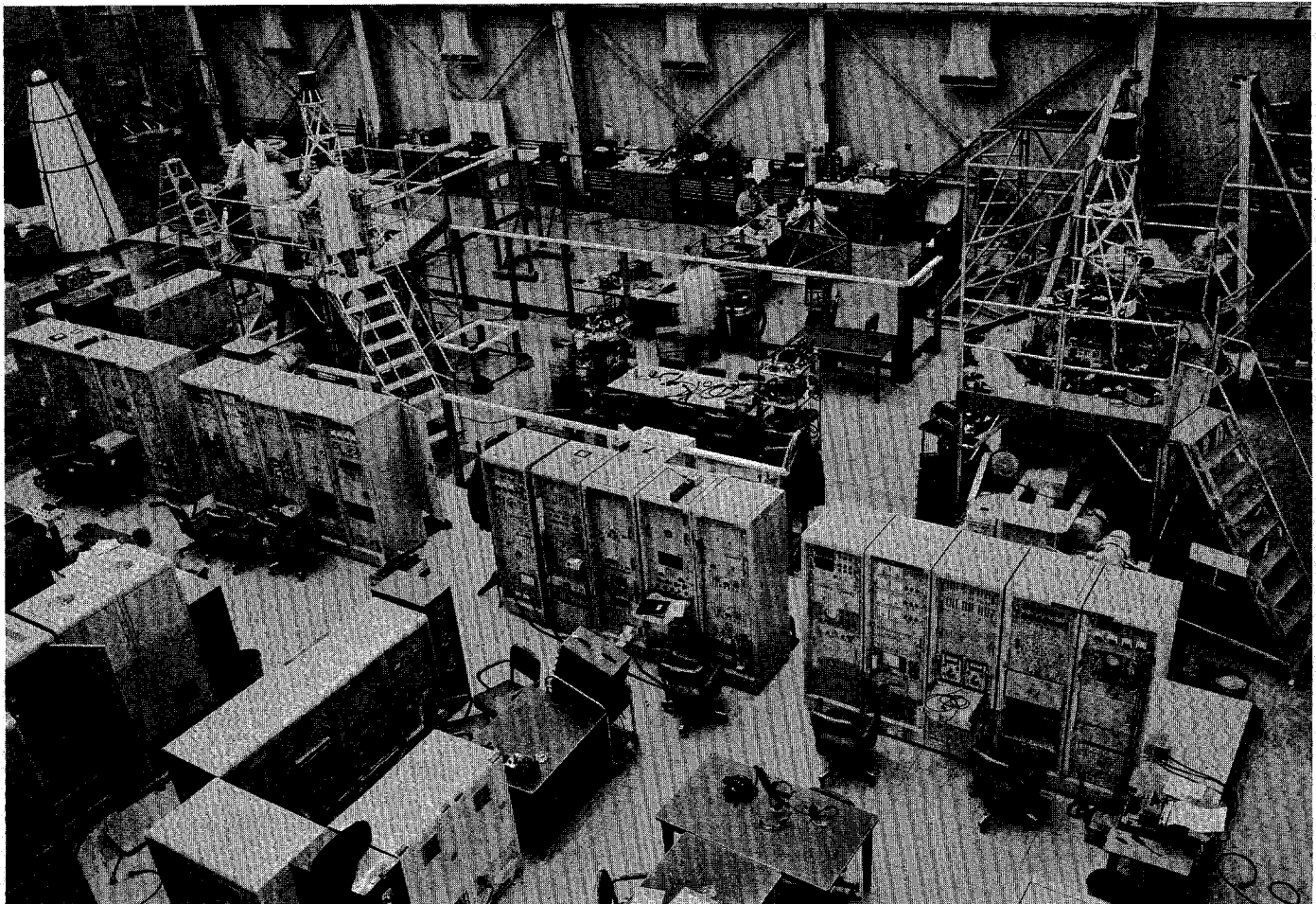


Figure 1. *Mariner* MR-1 and -2 test complex in spacecraft assembly facility

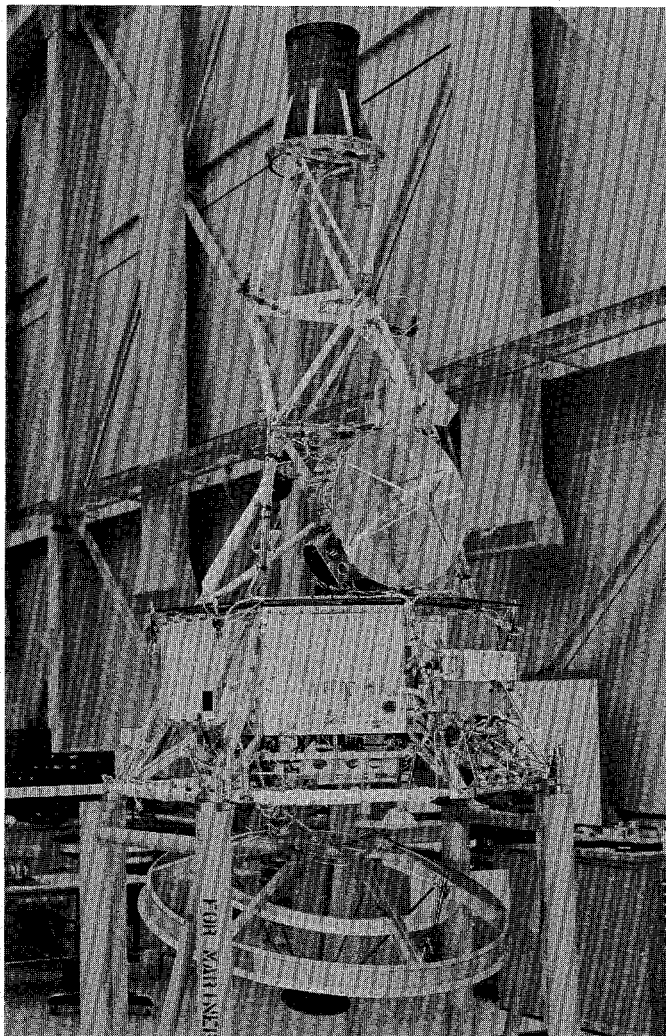


Figure 2. Mariner MR-1 in mechanical assembly area

angle and antenna hinge position measurements. These calibrations have not been completed due to the lack of a hinge actuator and output gear.

2. Mariner MR-2

a. Mechanical assembly. The mechanical assembly of MR-2 spacecraft started in late January 1962. A JPL adapter-torsion ring unit was utilized as an assembly platform for the spacecraft. The spacecraft hex frame was mounted on the JPL adapter torsion ring, and all modifications (i.e., drilling cable clamp holes, installing Davis nuts, cleaning and burnishing sharp edges, etc.) were performed. Subsequent to modification work, the gold-plated and polished aluminum surfaces of the hex frame and superstructure were wrapped with white flannel cloth to keep spacecraft clean during assembly, subsystem and system operations.

After completion of the preliminary hex frame and superstructure preparations, the spacecraft was assembled by installing the equipment in the following sequence: (1) ring harness, (2) mid-course propulsion system, (3) attitude control manifold plumbing and jet valves, (4) electronic cases (I through VI), (5) upper thermal shield, (6) omni-antenna, magnetometer, particle flux detector, and ion chamber, (7) cosmic dust detector, (8) superstructure, (9) all electrical and coax cables leading from the superstructure to the hex frame, (10) radiometer, (11) high gain antenna and Earth sensor, and (12) primary Sun sensors.

The spacecraft was then moved to the system test complex (STC) area for subsystem and system tests. Table 1 lists the scheduled tests and indicates those completed.

B. Telecommunications

1. Data Encoder Reliability Program

As part of the *Mariner R* data encoder program, a reliability program was initiated. The reliability program was designed to improve the probability of success of the flight data encoder. An initial analysis revealed several areas critical to reliability of the encoder, and specific tasks were established to deal with these areas. Particularly significant improvement in inherent reliability was obtained by:

- (1) Use of high reliability parts wherever possible.
- (2) Redesign of selected, critical portions of the system.

The opportunity to devise a new system configuration for *Mariner R* was used to make improvements in system reliability. The following presents an analysis of *Mariner R* encoder reliability and an estimate of its probability of success.

a. Performance requirements. The criterion for successful operation of the data encoder was defined as a minimum requirement of 90% probability of receiving 50% of the data through the encoder at Venus distance. The average ambient temperature of all parts is assumed to be 40°C.

b. System analysis. The data encoder block diagram is shown in Figure 3. For purposes of analysis, the system can be divided into two series-connected subsystems:

- (1) The basic system, which consists of those circuits that must operate without part failure for any data to be processed successfully.
- (2) The data input system, which includes everything outside the basic system.

Unlike the basic system, the data input system could sustain many part failures before the amount of data drops below the 50% criterion.

The probability P_s of encoder success is the product of the success probabilities of the two subsystems.

$$P_s = P_B \times P_I$$

where

P_B = the probability of success (failure-free operation) for the basic system

P_I = probability of success (at least half the channels operate successfully) for the data input system

Basic system. Probability of success may be estimated for the basic system by the usual reliability formula:

$$P_B = e^{-t \sum_{i=1}^N F_i} \quad (1)$$

where

F_i = failure rate of part i in its application

N = total number of parts in the basic system

t = time of system operation

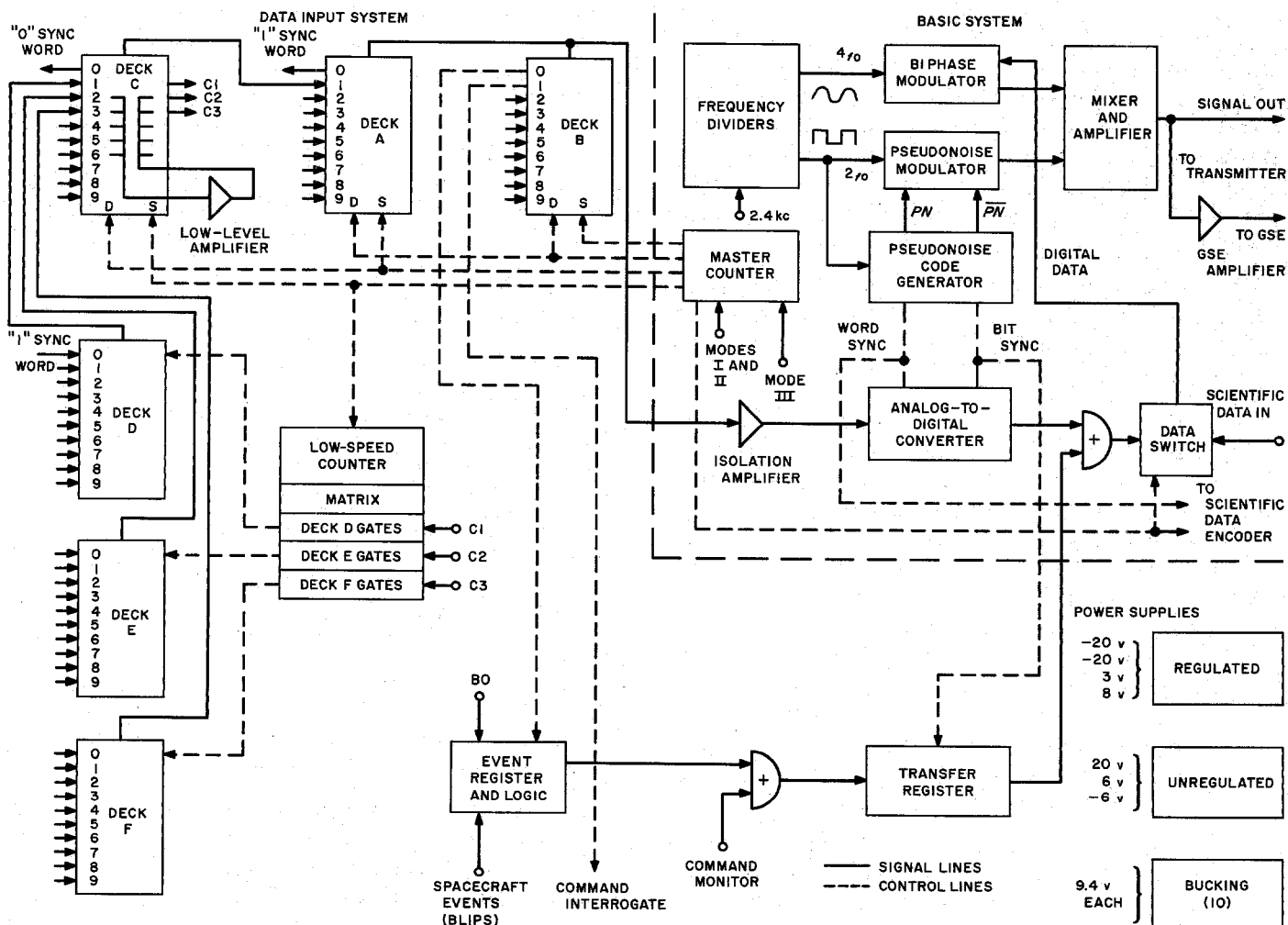


Figure 3. Mariner R data encoder

A failure rate was estimated for each part application using life test data and measurements of the actual electrical and environmental stresses imposed in the application. Estimated failure rates were then doubled so as to make allowances for unsuspected stresses and for stresses which are difficult to evaluate (e.g., unspecified radiation exposure). Table 2 lists the totals of part quantities, failure rates, and the expected number of failures for each circuit block and for the entire basic system. The failure rate of 10.446 per 10^6 hr of the basic system is doubled to obtain an estimated 20.892 per 10^6 hr. Earlier estimates, based on past counts and on basic failure rates without consideration for derating, indicated 165.445 per 10^6 hr for the original *Mariner A* design and 31.4 per 10^6 hr for *Mariner R*. Reliability of the basic system, based on this analysis, is estimated at 0.971 for 1460 hr in space flight. This computation is [using Equation (1)]:

$$P_B = e^{-1460(20.892 \times 10^{-6})}$$

$$= 0.971$$

It should be noted that *Basic System* portion of Table 2 includes the low-speed programmer in the basic system, although it is shown as belonging to the data input system in Figure 3. Since the switching decks operated by the programmer must supply some data to satisfy the system reliability criterion, the programmer must be included in the basic system as a part which cannot have a failure.

Data input system. Circuit block reliability figures are also given for the data input system, Table 2. The totals given would apply to system reliability if the data success criterion were 100%.

A reliability block diagram, exhibiting the operating dependencies within the data input system, is shown in Figure 4. This diagram shows that six ten-position switching decks (A through F) provide a total of 60 possible input positions. However, Positions A0, C0, and D0 are used for synchronizing signals, Positions B0 and B1 are used to actuate other data sources, and Positions C0, C1, C2, and C3 carry only signals from other decks. All 51 remaining positions are used for data input. To simplify the mathematical analysis, the Deck D sync word position D0 is treated as if it were a data position, and the total number of data input channels is taken as 52, so that the success criterion for 50% of the data is set as at least 26 successful data channels.

Each input can be traced on the diagram through its route to the basic system. The elements of the data input system which must operate successfully to deliver data over each route can be identified. Items critical to success for several routes are exposed. For example, failure

Table 2. Circuit block reliability

Basic system			
Circuit block	Total parts	Failure rate per 10^6 hr	Expected failures in 1460 hr
Power supply	100	1.3915	0.00407
Bucking supplies	80	0.9196	0.00266
Frequency dividers and filters	124	0.1667	0.00049
Mode and rate relay drivers	38	4.0202	0.01230
Pseudonoise generator	166	0.0725	0.00021
Biphase and pseudonoise modulators	71	1.4764	0.00432
Comparator isolation amplifier	26	0.1275	0.00037
Analog-to-digital converter	123	0.9893	0.00289
Analog-to-digital converter decoder	309	0.9666	0.00285
Data switch	16	0.0024	0.00001
Master counter	312	0.2368	0.00069
Low-speed programmer	329	0.0439	0.00013
GSE isolation amplifier	11	0.0135	0.00004
and gates, inverter, sync amplifier	34	0.0194	0.00006
Total	1,739	10.4463	0.03050
Data input system			
Blip registers	487	0.2812	0.00082
Transfer register	283	0.1636	0.00048
Command interrogate pulse amplifier	10	0.0042	0.00002
Antenna yoke temperature bridge	5	0.0250	0.00073
Low level amplifier	69	0.5312	0.00155
Deck A (one flip-flop, nine flip-flop switches)	313	0.3817	0.00111
Deck B (two flip-flops, eight flip-flop switches)	288	0.3504	0.00102
Deck C (one flip-flop, three switches, nine flip-flop switches)	418	0.4519	0.00132
Decks D, E, and F (30 switches)	570	0.7020	0.00240
Total	2,442	2.8912	0.00844
Universal flip-flop	16	0.0100	0.0000292
High-speed flip-flop	18	0.0217	0.0000634
Solid state switch	19	0.0234	0.0000681
Flip-flop-switch combination	37	0.0413	0.00012

of flip-flop A0 or A1 would result in loss of data from all but the eight input channels provided by Deck B, and thus would cause the encoder to fail to meet the reliability criterion. Failure of Switch S1, or of flip-flop C0 or C1, would lose all inputs but the sixteen obtainable from Decks A and B and encoder failure would occur.

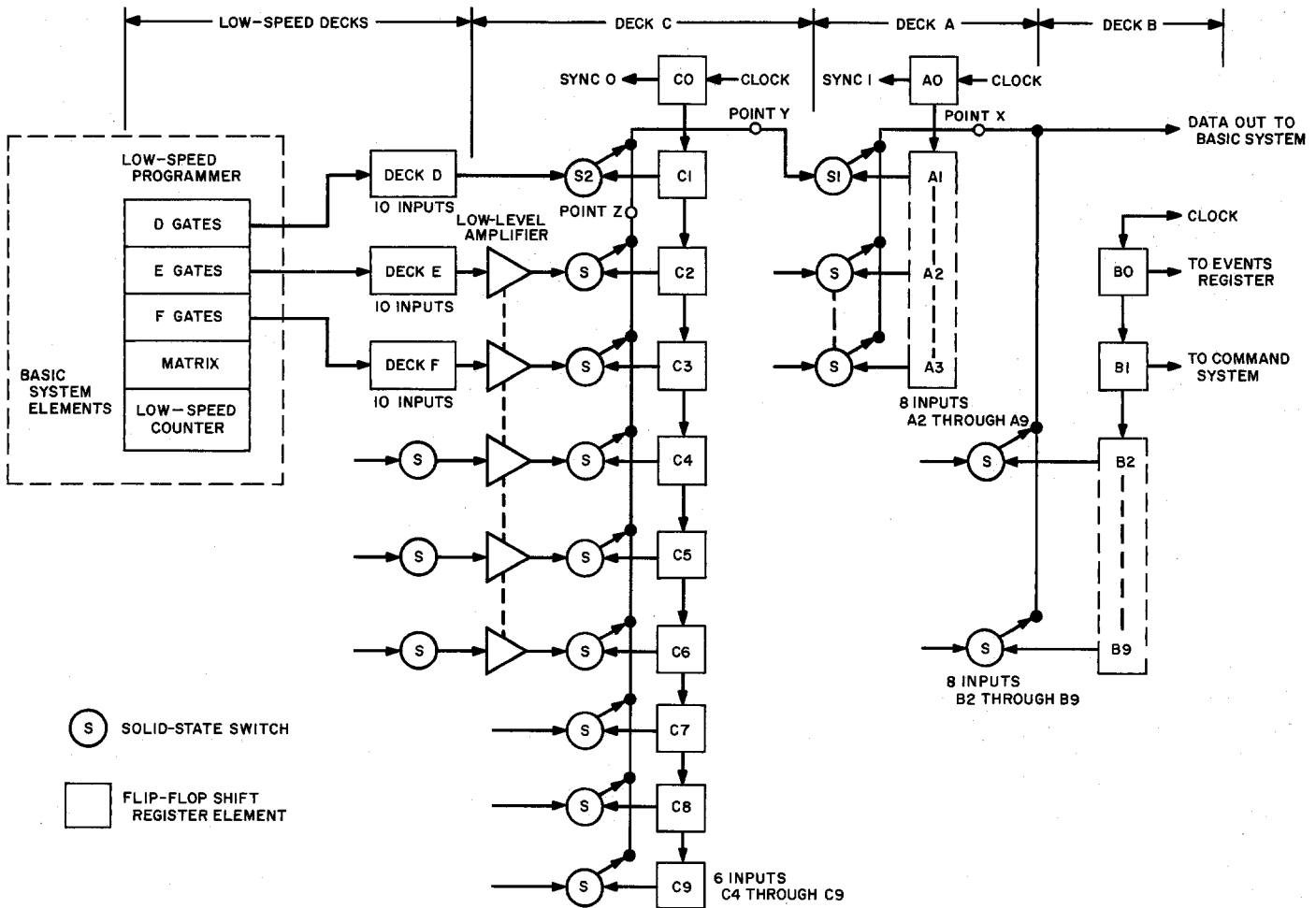


Figure 4. Mariner R reliability model of data encoder

A mathematical model for the probability of success, P_I of the data input system can be developed from analysis of the dependency relationships summarized in Figure 4. Thus, the probability of success is the sum, over possible values of i , of the probability $P_B(i)$ that i channels are delivered by Deck B multiplied by the probability $P(X \geq 26-i)$ that at least $26-i$ channels are delivered to Point X on the diagram. That is,

$$P_I = \sum_{i=0}^8 P_B(i) \cdot P(X \geq 26-i) \quad (2)$$

The probability $P(X \geq 26-i)$ that at least $26-i$ channels are delivered to Point X is the probability $P_{A0} \cdot P_{A1}$ that both A0 and A1 operate, multiplied by the sum, over possible values of j of the probability $P_A(j)$ that j channels are delivered to Point X from Deck A, times the probability $P(Y \geq 26-i-j)$ that at least $(26-i-j)$ channels are delivered to Point Y, times the probability P_{S1} that Switch S1 operates. Thus,

$$P(X \geq 26-i) = P_{A0} P_{A1} P_{S1} \sum_{j=0}^8 P_A(j) \cdot P(Y \geq 26-i-j)$$

and, by substitution in Equation (2),

$$P_I = P_{A0} P_{A1} P_{S1} \sum_{i=0}^8 \sum_{j=0}^8 P_B(i) P_A(j) P(Y \geq 26-i-j) \quad (3)$$

It is instructive to pause briefly at this point in the analysis and evaluate P_I for a special case. The probability $P(Y \geq 26-i-j)$ that at least $26-i-j$ channels are delivered to Point Y is the probability $P_{C0} \cdot P_{C1} \cdot P_{S2}$ that flip-flops C0 and C1 and Switch S2 will operate, multiplied by the sum, over possible values of m , of the probability $P_D(m)$ that m channels are delivered to S2 by Deck D times the probability $P(Z \geq 26-i-j-m)$ that at least $26-i-j-m$ channels are delivered to Point Z on the diagram. Or,

$$P(Y \geq 26-i-j) = P_{C0} \cdot P_{C1} \cdot P_{S2} \sum_{m=0}^{10} P_D(m) \cdot P(Z \geq 26-i-j-m)$$

and, substituting in Equation (3),

$$P_I = P_{A0} \cdot P_{A1} \cdot P_{C0} \cdot P_{C1} \cdot P_{S1} \cdot P_{S2} \\ \times \sum_{i=0}^8 \sum_{j=0}^8 \sum_{m=0}^{10} P_B(i) P_A(j) P_D(m) P(Z \geq 26 - i - j - m) \quad (4)$$

Consideration of the diagram and Equation (4) reveals that the minimum success criterion for the data input system can be met if all channels from Decks B1, A1, and D are delivered to the basic system, even if no data are presented to Point Z. Since this is the least complicated successful arrangement, the probability of its occurrence can be taken as a lower bound to the probability of success for the data input system. Fixing i , j , and m at their maximum value in Equation (4), the probability of input system success in this manner is given by:

$$P_I = P_{A0} \cdot P_{A1} \cdot P_{C0} \cdot P_{C1} \cdot P_{S1} \cdot P_{S2} \cdot \\ P_B(8) \cdot P_A(8) \cdot P_D(10) \cdot P(Z \geq 0) \quad (5)$$

Assuming that

$$P_{A0} = P_{A1} = P_{C0} = P_{C1}$$

since they are identical flip-flops, that

$$P_{S1} = P_{S2}$$

and that

$$P_B(8) = P_A(8) \cdot P_{A0}^2$$

and noting that $P(Z \geq 0) = 1$, then Equation (4) reduces to

$$P_I = P_{A0}^4 \cdot P_{S1}^2 \cdot P_A^2(8) \cdot P_{A0}^2 P_D(10)$$

or, combining terms, to

$$P_I = P_{A0}^6 P_{S1}^2 P_A^2(8) P_D(10) \quad (6)$$

Data examined earlier places the probability of success for each flip-flop, switch, or deck at 0.999 or higher. Consequently, a lower bound for the probability of success of the data input system can be taken as

$$P_I \geq (0.999)^{11} \approx 0.989$$

There are numerous successful channel combinations in addition to the one just evaluated. An estimate of the probability of success of the data input system must include the probabilities associated with all successful combinations. To develop such an estimate we return to development of the analysis, and in Equation (3), consider the term $P(Y \geq 26 - i - j)$. This term represents the probability that at least $(26 - i - j)$ channels are delivered to Point Y on the diagram from any combination of

sources in Decks C, D, E, and F. A maximum of 36 channels can be delivered to Point Y and success is assured if 26 channels or more are delivered, no matter how many are provided from Decks A and B. Thus, all combinations of individual channel successes providing a total of 26 or more channels to Point Y imply data input system success. In addition, since Decks A and B can supply up to 16 channels, combinations providing fewer than 26 (but at least 10) channels to Point Y can result in system success if they occur when $(i + j) > 0$.

It would be possible to identify successful states by enumerating all possible combinations of success and failure of elements of the data input system. Fortunately, this task is simplified by first grouping system elements into three sets:

- (1) All elements between Decks D, E, F, and Point Y.
- (2) All elements prior to (or leading to) and including C4.
- (3) Decks D, E, and F.

Consider first the elements between Decks D, E, and F, and Point Y. They provide three possible routes for data, one route per deck. The number of open routes determines the number of channels available Y from D, E, and F. Further, the integrity of portions of each route determines the number of additional channels which can be delivered from Deck C. Noting that at least one route must be open for system success, we first evaluate the probabilities P_{R3} , P_{R2} , P_{R1} that three, two, and one routes are open.

For all three routes to be open, four flip-flops (C0 to C3), three switches, and the amplifier must operate successfully. So, the probability of three open routes is given by

$$P_{R3} = a s^3 f^4$$

where a , s , and f are the probabilities of success for the amplifier, a switch, and a flip-flop, respectively.

The probability of exactly two open routes to Point Y is more complicated to evaluate. If the two open routes are for Decks D and E, then C0, C1, and C2, the switches associated with C1 and C2, and the amplifier must operate. In addition, C3 or its switch, or both, must fail. For this case

$$P_{R2} = a s^2 f^3 (1 - s f)$$

If the two open paths are for Decks D and F, it is necessary that four flip-flops (C0 to C3), two switches (those

for C1 and C3), and the amplifier operate and that the switch for C2 fail. For this case,

$$P_{R2} = as^2f^4(1-s)$$

The expression for the case in which the two open paths are for Decks E and F is identical.

The same procedure is used to enumerate the one-route cases and to compute the probability of occurrence for each. Also, it is used to enumerate and compute probabilities for cases wherein from zero to six additional channels from Deck C are delivered from Deck C Positions 4 through 9. It then is possible to define the probabilities $P_{R3}(n)$, $P_{R2}(n)$, $P_{R1}(n)$ that n channels are available from Deck C Positions 4 through 9 when three, two, and one routes are open for Decks D, E, and F. Table 3 lists the expressions derived for each value of n .

Table 3. Algebraic expressions^a for $P_{Rx}(n)$

$P_{R3}(0) = as^3f^4 \{ (1-f) [1+f(1-s)(1+s) + f^2(1-s)^2(1+s)^2 + f^3(1-s)^3 + f^4(1-s)^4(1+s)^3 + f^5(1-s)^5(1+s)^2] + f^6(1-s)^6(1+s)^3 \}$
$P_{R3}(1) = as^4f^5 \{ (1-f) [s + 2sf(1-s)(1+s) + 3sf^2(1-s)^2(1+s)^2 + f^3(1-s)^3(1+s)^2(4s+1) + f^4(1-s)^4(1+s)^2(5s+2)] + 3f^6(1-s)^5(1+s)^2(2s+1) \}$
$P_{R3}(2) = as^5f^6 \{ (1-f) [s^2 + 3sf(1-s)(1+s) + 3sf^2(1-s)^2(1+s)(2s+1) + f^3(1-s)^3(1+s)(3s^2+6s(1+s)+(1+s)^2)] + 3f^4(1-s)^4(1+s)[s^2+3s(1+s)+(1+s)^2] \}$
$P_{R3}(3) = as^6f^7 \{ s(1-f)[s^2+sf(1-s)(4s+3) + f^2(1-s)^2(s^2+6s(1+s)+3(1+s)^2)] + f^3(1-s)^3[s^3+9s^2(1+s)+9s(1+s)^2(1+s)^2] \}$
$P_{R3}(4) = as^8f^8 \{ s(1-f)[s+f(1-s)(5s+3) + f^2(1-s)^2[3s^2+9s(1+s)+3(1+s)^2]] \}$
$P_{R3}(5) = as^{10}f^9 \{ s(1-f) + 3f(1-s)[2s+1] \}$
$P_{R3}(6) = as^{12}f^{10}$
$P_{R2}(0) = P_{R3}(0)[3s^{-1}(1-s)] + as^2f^3(1-f)$
$P_{R2}(1) = P_{R3}(1)[3s^{-1}(1-s)] \quad i = 1, 2, \dots, 6$
$P_{R1}(0) = P_{R3}(0)[3s^{-2}(1-s)^2] + s^2f^2(1-f)[1+2f(1-s)]$
$P_{R1}(1) = P_{R3}(1)[3s^{-2}(1-s)^2] + s^2f^3(1-a)\{(1-f)[1+3f(1-s)] + 3f^2(1-s)^2\}$
$P_{R1}(2) = P_{R3}(2)[3s^{-2}(1-s)^2] + s^3f^3(1-a)[(1-f) + 3f(1-s)]$
$P_{R1}(3) = P_{R3}(3)[3s^{-2}(1-s)^2] + s^4f^4(1-a)$
$P_{R1}(i) = P_{R3}(i)[3s^{-2}(1-s)^2] \quad i = 4, 5, 6$

^aProbability $P_{Rx}(n)$ that n channels are available from Deck C positions 4 through 9 when x routes are open from Decks D, E, and F to Y.

To complete the evaluation of $P(Y \geq 26-i-j)$ the probability $P_{DEF}(m)$ of m channels being available for routing to Point Y from Decks D, E, and F is given by the binomial term for the probability of m successes out of x . That is,

$$P_{DEF}(m) = (m^x) P^m q^{x-m} \quad 0 \leq m \leq x$$

Since each channel success in Decks D, E, or F depends upon operation of a single switch, then

$$P_{DEF}(m) = (m^x) S^m (1-s)^{x-m} \quad 0 \leq m \leq x$$

An expression can now be written for the probability that at least $(26-i-j)$ channels are delivered to Point Y.

$$P(Y \geq 26-i-j) = \sum_{K=26-i-j}^{36} \sum_{m=K-6}^K \times [(m^{30}) s^m (1-s)^{30-m} P_{R3}(K-m) + (m^{20}) s^m (1-s)^{20-m} P_{R2}(K-m) + (m^{10}) s^m (1-s)^{10-m} P_{R1}(K-m)] \quad (7)$$

where

$$(m^x) = 0 \quad \text{if } x < m$$

The first term within brackets represents the probability of success when three routes are open from Decks D, E, and F to Point Y. The second term is associated with the two-route and third term with the one-route condition.

Substituting Equation (7) in Equation (3)

$$P_I = P_{A0} \cdot P_{A1} \cdot P_{N1} \sum_{i=0}^8 \sum_{j=0}^8 \sum_{K=26-i-j}^{36} \sum_{m=K-6}^K P_B(i) P_A(j) \times [(m^{30}) s^m (1-s)^{30-m} P_{R3}(K-m) + (m^{20}) s^m (1-s)^{20-m} P_{R2}(K-m) + (m^{10}) s^m (1-s)^{10-m} P_{R1}(K-m)] \quad (8)$$

where

$$(m^x) = \frac{X!}{m!(x-m)!}$$

if $0 \leq m \leq x$ and

$$(m^x) = 0 \quad \text{if } x < m$$

$P_B(i)$ and $P_A(j)$ for specific values of i and of j are given in Table 4. In Table 4 $P_A(j)$ includes P_{A0} , P_{A1} , and P_{S1} .

An IBM 1620 computer program has been written to evaluate data input system reliability by Equation (8). It

Table 4. Reliability^a terms for Decks A and B

$P_B(0) = 1 - s f^8 [1 - f^8 (1 - s)^8] [1 - f(1 - s)]^{-1}$
$P_B(1) = s f^7 \left\{ (1 - f) \left[\sum_{i=1}^7 \left(\frac{i}{1} \right) f^{i-1} (1 - s)^{i-1} \right] + 8 f^7 (1 - s)^7 \right\}$
$P_B(2) = s^2 f^6 \left\{ (1 - f) \left[\sum_{i=1}^6 \left(\frac{i+1}{2} \right) f^{i-1} (1 - s)^{i-1} \right] + 28 f^6 (1 - s)^6 \right\}$
$P_B(3) = s^3 f^5 \left\{ (1 - f) \left[\sum_{i=1}^5 \left(\frac{i+2}{3} \right) f^{i-1} (1 - s)^{i-1} \right] + 56 f^5 (1 - s)^5 \right\}$
$P_B(4) = s^4 f^4 \left\{ (1 - f) \left[\sum_{i=1}^4 \left(\frac{i+3}{4} \right) f^{i-1} (1 - s)^{i-1} \right] + 70 f^4 (1 - s)^4 \right\}$
$P_B(5) = s^5 f^3 \left\{ (1 - f) \left[\sum_{i=1}^3 \left(\frac{i+4}{5} \right) f^{i-1} (1 - s)^{i-1} \right] + 56 f^3 (1 - s)^3 \right\}$
$P_B(6) = s^6 f^2 \left\{ (1 - f) \left[\sum_{i=1}^2 \left(\frac{i+5}{6} \right) f^{i-1} (1 - s)^{i-1} \right] + 28 f^2 (1 - s)^2 \right\}$
$P_B(7) = s^7 f [(1 - f) + 8 f (1 - s)]$
$P_B(8) = s^8 f^0$
$P_A(0) = s [P_B(0) + f^8 - 1]$
$P_A(i) = s P_B(i) \quad i = 1, 2, \dots, 8$
^a Probability of i channels from Decks A and B.

requires as input only the following three reliabilities (with the values obtained from their stress analysis failure rates)

$$f, \text{ flip-flops,} = 0.99997$$

$$s, \text{ switches,} = 0.99993$$

$$a, \text{ low-level amplifier,} = 0.99845$$

Using these values, the computer program estimates the reliability of the data input system to be

$$P_I = 0.99981$$

Composite encoder reliability. The product of the two figures obtained in the foregoing analyses is the reliability estimate for the complete data encoder. Thus

$$P_S = P_B P_I = (0.971)(0.9998) = 0.9708$$

or

$$P_S = 97.08\%$$

With the total failure rates assembled in Table 2, it is now of interest to consider the total system reliability for 100% of the data. The total estimated failure rates amount to $10.446 + 2.8912 = 13.375$ per 10^6 hr. This is doubled to account for unexpected stresses and the estimated total is 26.675 per 10^6 hr. P_S is computed using the usual reliability formula [expressed in Equation (1)]

$$\begin{aligned} P_S &= e^{-1460(26.675 \times 10^{-6})} \\ &= e^{-0.0390} \\ &= 0.96175 = 96.175\% \end{aligned}$$

c. Conclusion and recommendations. The degree of success of the *Mariner* reliability improvement program will best be measured by the performance of the data encoders operating for many hours in system test, final checkouts, and ultimately on their deep space mission.

The reliability improvements indicated by this analysis have resulted from the use of highly reliable parts, subjected to conservative stresses, in proven designs. The reliability estimate given here is the best available evaluation of the inherent design reliability. Many controls and checks are required and applied in the manufacturing process to assure the realization of this reliability in each data encoder. The control phase of the reliability program embraces 100% inspection and testing of all parts, careful handling of parts and assemblies, stringent workmanship standards, and a clean working area. Part discrepancies and failures are analyzed, and corrective action is taken where necessary. Performance records are maintained on each system until launch. The principal recommendation deriving from this reliability program is that reliability effort on such a system must begin with the development of system concepts in order for its maximum effectiveness to be realized.

C. Guidance and Control

1. Autopilot

a. Description of configuration. The mid-course autopilot controls the spacecraft during the firing of the mid-course rocket. The configuration of the pitch and yaw autopilot channels is shown in Figure 5, the roll channel in Figure 6, and the combined three channel autopilot in Figure 7.

The signal for the autopilot is generated by a rate gyro in a closed electronic loop. The torquer current of the gyro produces a voltage which is a measure of the angular rate, and passing the torquer current through a capacitor produce a voltage (across the capacitor) which is a measure of angular position. The combined rate plus position signal is the output of the gyro loop. The signals from the three gyro loops are combined to drive four jet vanes, which are located in the exhaust stream of the rocket and provide pitch, yaw, and roll torques. (The roll axis is the axis of symmetry of the spacecraft; the pitch axis passes through the solar panels.)

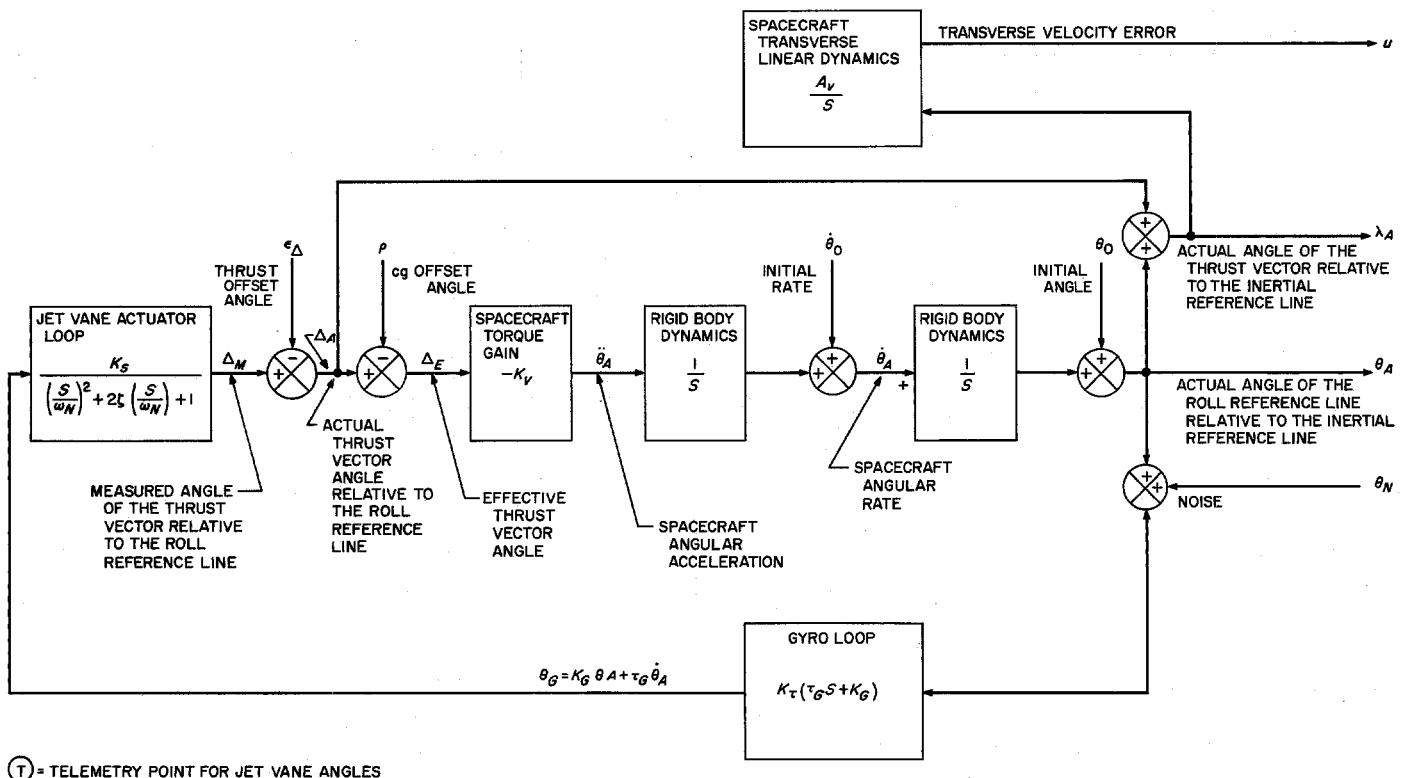


Figure 5. Pitch or yaw autopilot functional model diagram

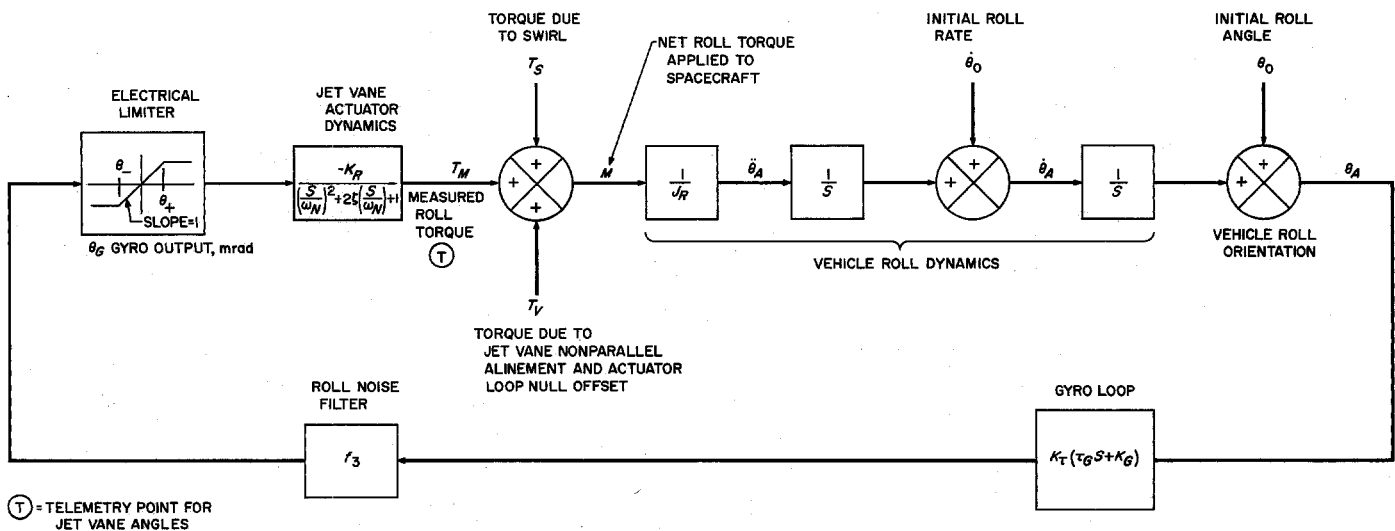


Figure 6. Roll autopilot functional diagram

The loop gain setting of each autopilot channel is based on the criterion that it must be as low as possible with the limitation that the peak signal resulting from the worst starting transient must not saturate the gyro loop. The purpose of setting the gain as low as possible is to avoid, to as great an extent as possible, interactions

between the autopilot and elastically coupled structural members, such as the solar panels and the high gain antenna. However, the configuration of the gyro loop results in signal saturation limitations which do not allow the cutoff frequency of the autopilot to be set low enough to completely avoid these problems.

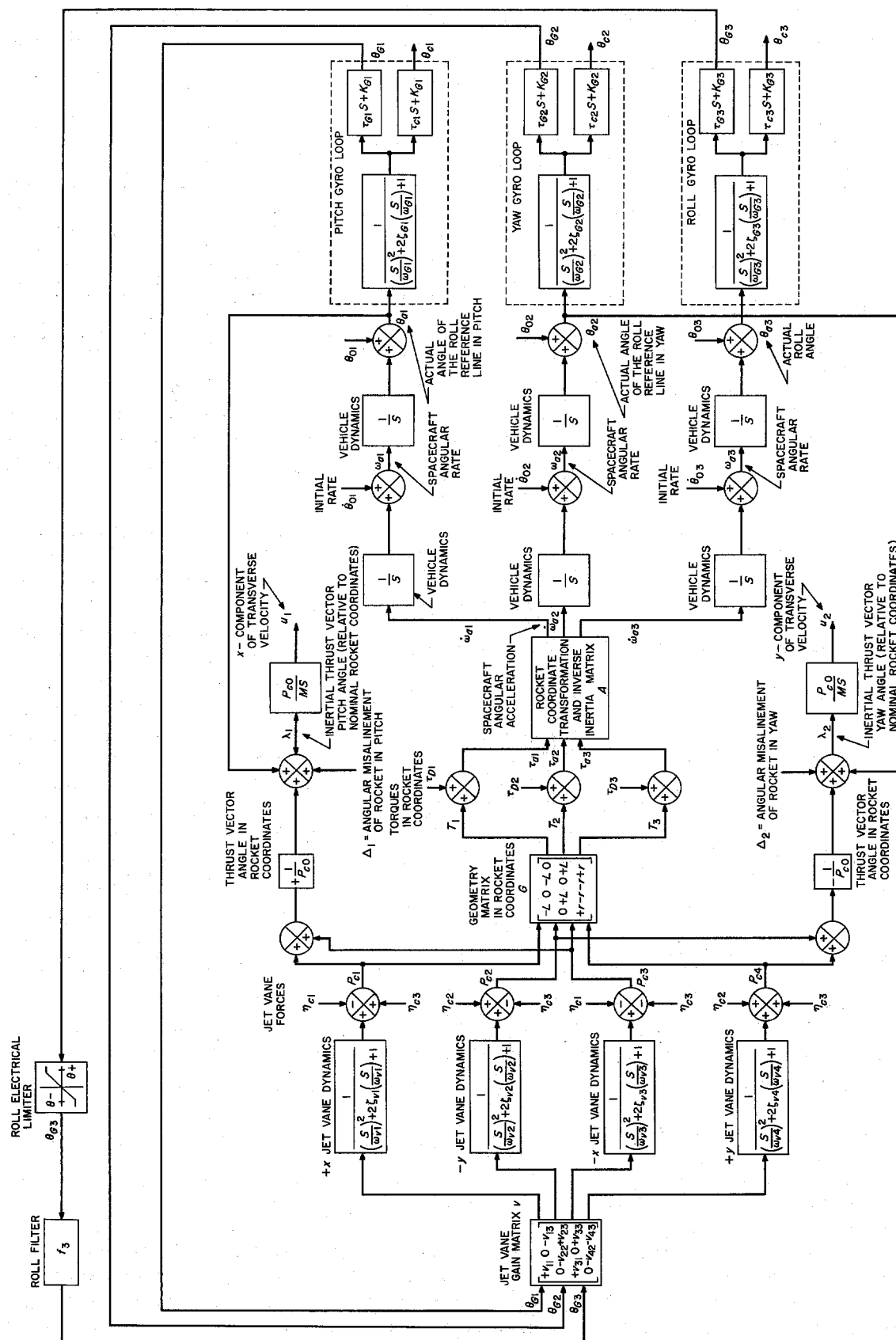


Figure 7. Three-axis autopilot functional diagram

v = STATIC GAINS OF JET VANES lb-in/rad of gyro output
 L = DISTANCE FROM JET VANE CENTER OF LIFT TO CG
 r = RADIAL DISTANCE FROM ROCKET CG TO CENTER OF LIFT OF THE JET VANES
 A = THE PRODUCT OF THE COORDINATE TRANSFORMATION FROM ROCKET TO SPACECRAFT COORDINATES, C , AND THE INVERSE INERTIA TENSOR, J^{-1} , $A = J^{-1}C$
 P_{e0} = NOMINAL ROCKET THRUST
 M = SPACECRAFT MASS
 Δ_1 = ANGULAR MISALIGNMENT OF ROCKET IN PITCH
 Δ_2 = ANGULAR MISALIGNMENT OF ROCKET IN YAW
 λ_1 = THRUST VECTOR ANGLE IN ROCKET COORDINATES
 λ_2 = THRUST VECTOR YAW ANGLE (RELATIVE TO NOMINAL ROCKET COORDINATES)
 $\omega_{e1}, \omega_{e2}, \omega_{e3}$ = SPACECRAFT ANGLE OF THE ROLL REFERENCE LINE IN PITCH, YAW, ROLL; 1, 2, 3, 4 = JET VANE LOCATIONS $+x, +y, -x, -y$ (NEW CONVENTION)

(For a more complete description of the autopilot, see SPS 36-7, pp 24-31, and SPS 37-12, p 17, which describe the *Ranger III* autopilot. The *Mariner R* autopilot is similar to *Ranger III* except that *Mariner R* guidance accuracy does not require a path guidance loop for accurate control of the thrust-vector inertial direction.)

b. Autopilot interactions with elastically coupled members. Analysis of the interactions of the autopilot with elastically coupled members includes the following possible problem areas:

- (1) Solar panels; two bending modes in yaw, one in roll, and the torsional mode in pitch.
- (2) High gain antenna; one mode about each of three axes.
- (3) Superstructure.
- (4) Radiometer.
- (5) Fuel slosh.

The stability analysis of the autopilot, as affected by the elastically coupled structural members, is being approached in three areas. These are:

- (1) The effects of the various resonances have been analyzed separately in one channel of the autopilot at a time. Here each mode of each elastically coupled member is analyzed individually as though it acted independently of other modes and other elastically coupled members.
- (2) A digital computer is being programmed to determine the roots of the characteristic equation for the six-channel autopilot (Fig 8) with elastically coupled members. The values of the real parts of these roots are a measure of system stability.
- (3) An analog computer is being used to simulate a linear representation of the elastically coupled members acting together. The block diagram for this simulation is Figure 8. It will be noted that this drawing is an extension of the three-channel autopilot block diagram of Figure 7. In the set of loops designated by the subscript i , there is one loop for each elastically coupled member. The impedance matrices Z_i are 6 by 6 matrices whose elements are ratios of second order quadratics in the complex frequency (Laplace operator) s .

Analysis of individual resonant modes. As stated above, the first approach taken to accomplish stability is to analyze each mode of each elastically coupled member as though it acted independently of other modes and

other elastically coupled members. This is an assumption that yields optimistic results. However, the modal properties were estimated conservatively to compensate for this effect.

Stability of the autopilot in the face of a single elastic-coupling (structural) resonance can be verified by either one of two criteria. First, stability is confirmed if the resonant frequency is sufficiently far removed from frequency regions where the rigid-body autopilot (designed for a spacecraft with no elastically coupled members) has between 180 and 360-deg phase shift. These regions are called the "elastically critical frequency regions." The second criterion, which must be used when the first is not met, requires that the damping ratio of the resonance must be large enough to make the loop gain of the autopilot less than unity at the resonant frequency. The measurable structural resonances for *Mariner R* have been investigated; in each case at least one of the criteria is satisfied.

Derivation of both criteria can be made directly from a typical pole-zero plot (Fig 9). The effect of an elastically-coupled member on the pole-zero plot is to modify the rigid body pole-zero plot by the addition of the two resonant pole-zero combinations near the imaginary axis, as shown. The zero occurs at the complex frequency for which there is no angular response of the spacecraft to autopilot torques, which is (ignoring translation) the resonant frequency of the elastically-coupled member when the spacecraft is held fixed. The pole occurs at the complex frequency (which is the resonant frequency of the coupled member when the spacecraft is free) where it requires no external torque to maintain angular response of the spacecraft. For the free case, conservation of angular momentum requires the spacecraft to turn in opposition to the elastically coupled member; hence, the resonant frequency is higher than in the constrained-spacecraft case. Consequently, the pole is at a higher frequency than the zero.

In deriving the stability criteria, the following conditions are required:

- (1) The separation of the pole and the zero is large compared with the distance of either from the imaginary axis of the complex-frequency (s) plane.
- (2) The separation of the pole and the zero is small compared with the distance to the origin.

Critical frequency region and stability criterion. The existence of the critical frequency region will be demonstrated in the following; root locus properties are used to indicate where the boundaries of the critical frequency

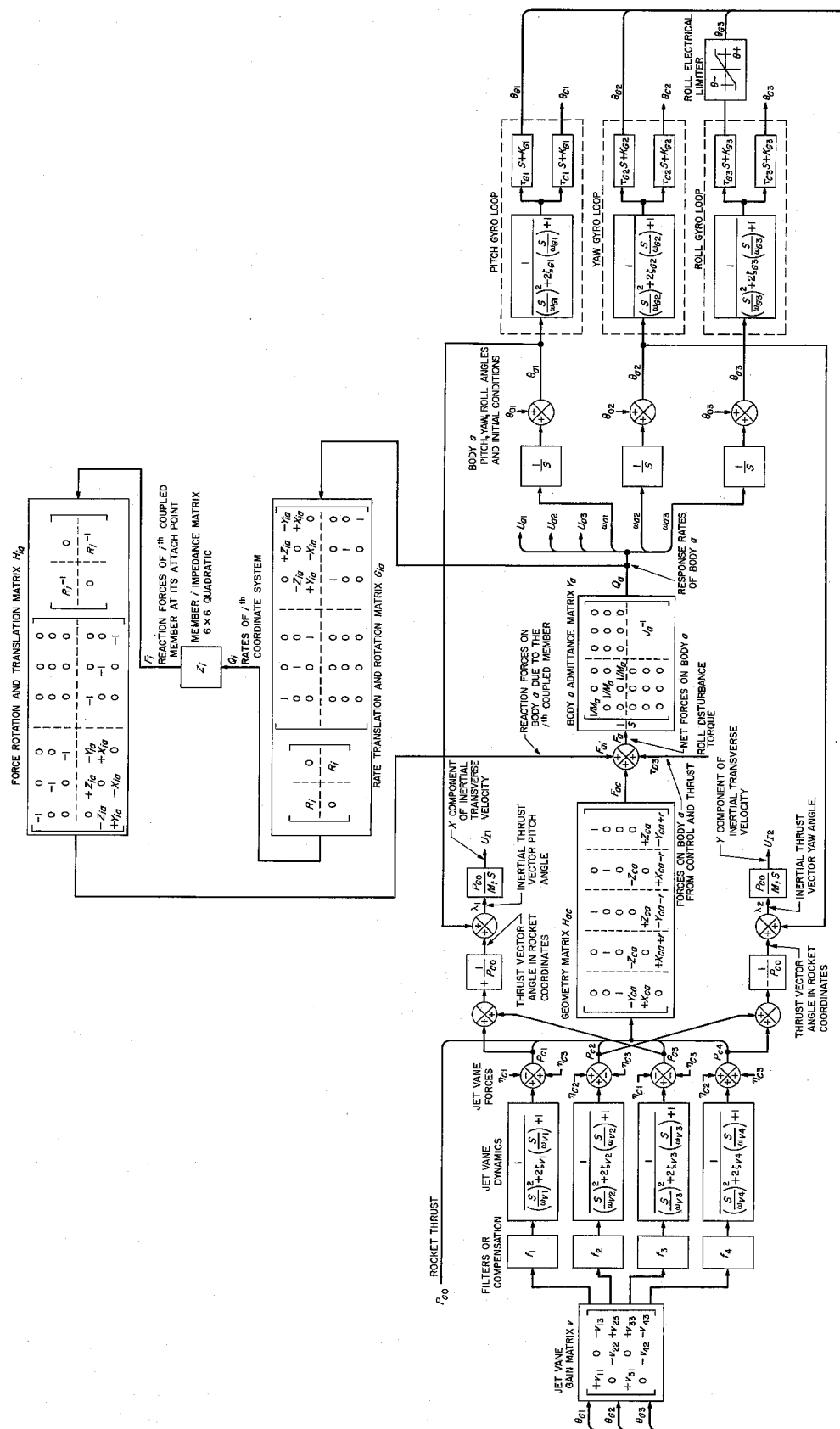


Figure 8. Six-channel autopilot with elastically coupled structural members

ELEMENTS OF THE γ MATRIX ARE STATIC GAINS OF JET VANES IN lb-in/ft² gyro output!
 $\gamma_{\text{c}20}, \gamma_{\text{c}21}, \gamma_{\text{c}22}$ = COORDINATES OF A POINT ON ζ OF THE ROCKET AT THE CENTER-OF-LIFT OF JET VANES;
 THESE COORDINATES ARE EXPRESSED IN A COORDINATE SYSTEM HAVING ITS ORIGIN AT THE CENTER-OF-GRAVITY OF BODY a
 $M_{\text{c}20}$ = MASS OF BODY a AND $J_{\text{c}20}^2 = I_{\text{c}20}^2$ IS ITS INVERSE INERTIA TENSOR
 $\gamma_{\text{c}20}^2, \gamma_{\text{c}21}^2, \gamma_{\text{c}22}^2$ = 1, 2, 3, 4, 5, 6 = PITCH, YAW ROLL; 1, 2, 3, 4 = JET VANS LOCATED THIS: $+x_1, -x_1, -y_1$ (NEW CONVENTION); $\gamma = a, b, c, d$, ETC. a = BODY a (CENTRAL BODY), b = MEMBER b , c = MEMBER c , ETC.
 ONLY THE EFFECTS OF THE j^{th} MEMBER ARE SHOWN IN GENERAL THERE WILL BE SEVERAL ELASTICALLY COUPLED MEMBERS
 $\gamma_{\text{c}20}^2$ = 3x3 ROTATION MATRIX OF THE COORDINATE SYSTEM OF THE j^{th} COUPLED MEMBER
 $\gamma_{\text{c}20}^2$ VECTORS ARE $(\gamma_{\text{c}20}^2)_1, (\gamma_{\text{c}20}^2)_2, \gamma_{\text{c}20}^2$, WHERE P = TRANSLATION FORCES, AND τ = TORQUES
 $\gamma_{\text{c}20}^2$ VECTORS ARE $(\gamma_{\text{c}20}^2)_1, (\gamma_{\text{c}20}^2)_2, \gamma_{\text{c}20}^2$, WHERE P = TRANSLATION VELOCITIES, AND ω = ANGULAR VELOCITIES
 M_j = TOTAL MASS OF THE SPACECRAFT AND ALL ITS MEMBERS
 $M_j^{-1} = 1/M_j$ IS THE INVERSE INERTIA TENSOR OF BODY a

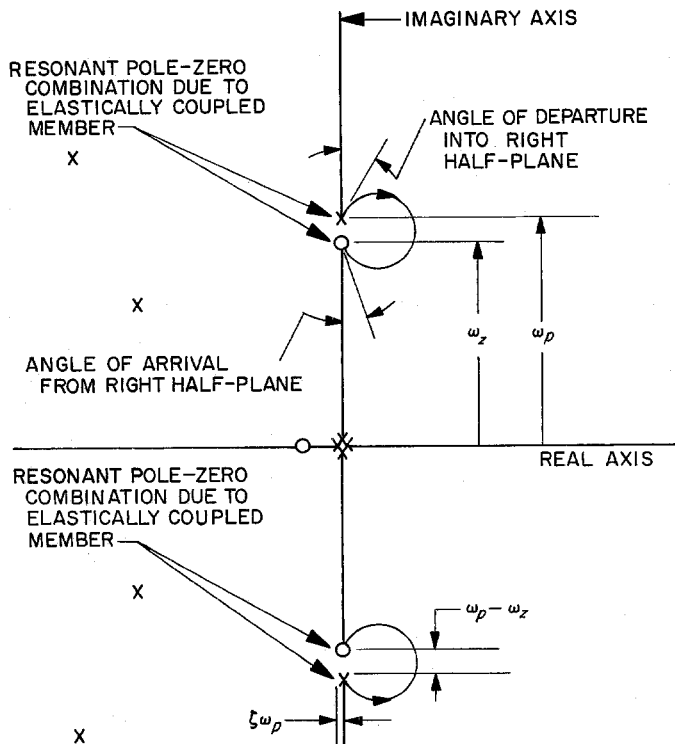


Figure 9. Typical autopilot pole-zero plot showing one elastically coupled member

region lie. A stability criterion is derived for resonances that are within the critical frequency region and is based on the open loop pole-zero plot.

If the added resonant pole and zero were undamped, they would lie on the imaginary axis. The angle of departure of the root locus from the pole into the right half-plane (Fig 9) is equal to the amount by which the phase shift of the rigid body autopilot exceeds 180 deg at the frequency of the pole. Negative angles imply departure into the left half-plane. The angle of arrival of the root locus at the zero from the right half-plane is equal to the amount by which the rigid body autopilot phase shift exceeds 180 deg at the frequency of the zero. Thus, if at the pole frequency the autopilot phase shift is 200 deg without including the effects of the elastically-coupled members, then the root locus will depart from the pole at an angle of 20 deg into the right half-plane. This indicates that there is a critical frequency region, where the rigid body autopilot phase shift is between 180 and 360 deg. Any resonances within this region must be examined to be sure that the damping ratio is large enough to avoid instability. Furthermore, it is not sufficient for the pole to be barely out of this region because, although the root locus initially departs into the left half-plane, it will cross into the right half-plane for a small

value of gain if the phase shift of the rigid-body autopilot is nearly 180 deg.

The autopilot may still be stable even though the resonant frequencies of elastically-coupled members are in the critical frequency region, provided that there is enough damping to keep the maximum loop gain in the region of resonance smaller than unity. This fact can be easily translated into a mathematical stability criterion as follows: If $G_R(S)$ is the rigid-body autopilot loop gain, then in the neighborhood of the pole, the loop gain $G_E(S)$ of the elastically-coupled autopilot is obtained directly from the pole-zero plot by adjusting G_R for the effects of the resonant pole-zero combinations that have been added.

This can be done by the use of vectors which extend from these added poles and zeros to the point on the imaginary axis representing the frequency of interest. The rigid body gain G_R is multiplied by the product of vectors from the added zeros, and divided by the product of vectors from the added poles. In this case the maximum value of the resulting gain is of interest, so the point of concern is on the imaginary axis opposite either of the added poles.

$$G_E(\omega_p) = G_R(\omega_p) \frac{\omega_p - \omega_z}{\zeta \omega_p}$$

For stability G_E must be less than unit. Thus

$$\frac{G_R(\omega_p)}{\zeta} \left(\frac{\omega_z}{\omega_p} - 1 \right) < 1$$

But for the open loop gain at very low and very high frequency to be properly related to the locations of the resonant pole-zero combination, it is necessary that

$$\frac{\omega_z}{\omega_p} = \left(\frac{J_L}{J_H} \right)^{1/2}$$

where J_L is the spacecraft moment of inertia at low frequency (below resonance) and J_H is the moment of inertia at high frequency (above resonance). The stability criterion is then

$$\frac{G_R(\omega_p)}{\zeta} \left[\left(\frac{J_L}{J_H} \right)^{1/2} - 1 \right] < 1$$

Any independent resonance which either lies well outside of the critical-frequency region or which meets the above damping criterion will not by itself cause the autopilot to be unstable.

2. Antenna Drive and Autopilot Electronics

The antenna electronics module contains an on-off, controlled dead-band amplifier which drives the actuator in response to either an internally stored reference signal or the Earth sensor command signal. The module also contains an automatic reference update system which can be ground controlled to insert the initial reference angle before launch or to pulse the antenna in flight as an emergency backup measure. Testers have been fabricated for rapid checkout of both modules.

The autopilot electronics contains the four power amplifiers for the jet vane actuators and the necessary circuitry to mix the pitch, yaw, and roll signals.

Four antenna and autopilot electronic subassemblies have been fabricated, functionally tested, flight acceptance tested, and delivered for system integration.

The *Mariner R* flight acceptance procedure consists of a high frequency, complex wave vibration test and vacuum-temperature test. In the vibration test the assembly is loaded in such a manner as to make it dynamically similar to the flight configuration. The assembly is subjected to the vibration test in a direction parallel to the vehicle axis of thrust and in two other orthogonal directions which are perpendicular to the axis of thrust. The test consists of a programmed sequence of band-limited white Gaussian noise and combined noise and sinusoidal vibration. The total time in each orthogonal direction is three minutes.

In the vacuum-temperature test the assembly is placed in a vacuum chamber at a pressure of 10^{-4} mm Hg or less. The temperature is controlled by cold wall and a bank of heat lamps, and then is measured at the largest internal mass. The module must operate properly for 1 hr at a temperature of 0°C and 2 hr at 55°C . The module must function as required by the test specification after completion of all tests.

3. Sun Simulator

Progress has been made toward the design of a true Sun simulator adequate for the testing of accurate Sun sensors.

One difficult Sun simulation problem has been the change in brightness across the xenon arc lamp source. To reduce this differential, only the brightest portion of the arc adjacent to the cathode is utilized. Although the diameter of this spot is less than 1 mm, the brightness from the upper to the lower edge varies by a factor of two. If uncorrected this effect would create a simu-

lated Sun of nonuniform brightness, which would introduce large pointing errors when Sun sensors (which seek the radiant center), were tested.

There are two apparent solutions to this problem. One is the utilization of a light pipe scrambler upstream from the aperture as shown in Figure 10.

The light pipe would consist of a cylindrical rod of quartz having flat polished ends. Radiation would exit from the rod end at the same cone angle that it entered, assuming no errors in the geometry of the rod. However, the internal reflections acting on a given ray will cause it to exit at a random circular angle dependent on the length and diameter of the rod. It is this characteristic that causes the scrambler to integrate the energy received and present a more uniform intensity at its output end.

The second solution to this problem involves the use of fiber optics. In this case the scrambler would consist of an incoherent fiber optics bundle. This scrambler would receive energy from a given point or fiber at one end, and transmit this energy to a random point at its output end. The effectiveness of this device would depend primarily upon the diameter of the fibers and how well they could be placed in random orientation at one end compared to the other.

A contract has been let for the design and fabrication of an optical system utilizing the light pipe scrambler first described. These optics will be used in conjunction with a 5000-w xenon arc lamp for the construction of the Mod II Sun simulator.

As a backup design the fiber optics bundle is being considered, with the mechanical design and construction being done at JPL. The Mod I simulator using standard optical components and a 900-w xenon lamp is also being designed and constructed at JPL. This is a portable unit designed to energize Sun sensors while they are being environmentally tested as well as to test their output characteristics at ambient conditions.

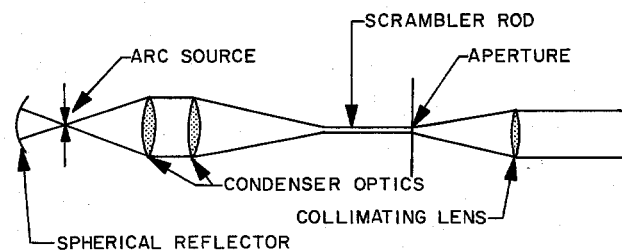


Figure 10. Sun simulator utilizing light-pipe scrambler

4. Long Range Earth Sensor

The long range Earth sensor (LRES) used on *Mariner R* uses a single photomultiplier tube together with a lens, a fixed aperture, and a modulating mask. The modulating mask is attached to a vibrating reed with a frequency of approximately 22 cps. The electronics associated with the reed (Fig 11) and the reed itself operate closed loop with a position pickoff device such that the reed oscillates at a constant amplitude. Frequency of oscillation is determined by the mechanical resonance of the reed only. Figure 12 shows the relative position of the optical elements. The knife edges on the modulating mask are in the focal plane of the optical system. The photomultiplier tube serves only as a light detector and has no position discrimination by itself.

The vibrating reed moves through a sufficient arc to completely uncover the fixed aperture at the extremes of motion. The output, therefore, is a series of pulses. A shift in the position of the Earth image in hinge causes a variation in the output pulse width. Similarly, a shift in roll causes a variation in the phase (or time) of the output relative to the reed motion. The electronics senses these properties in the pulse chain and at the output produces a dc voltage proportional to this error. Also present at the output are signals which indicate that an object is being tracked by the sensor and the amount of light being sensed.

The threshold of the unit is approximately 5×10^{-11} w/cm² (bolometric at a color temperature of 6000°K). The maximum intensity is a function of photomultiplier

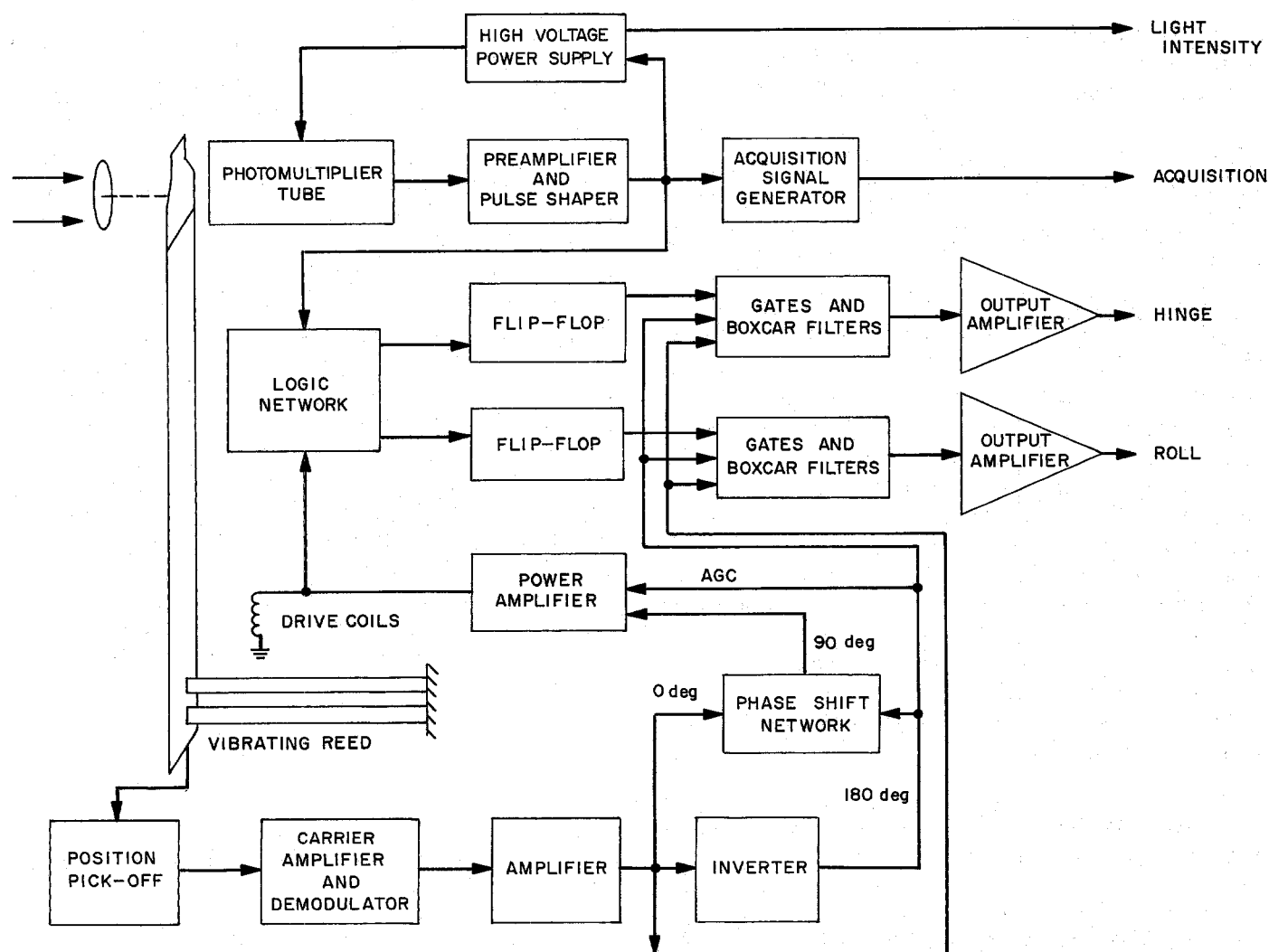


Figure 11. LRES circuit

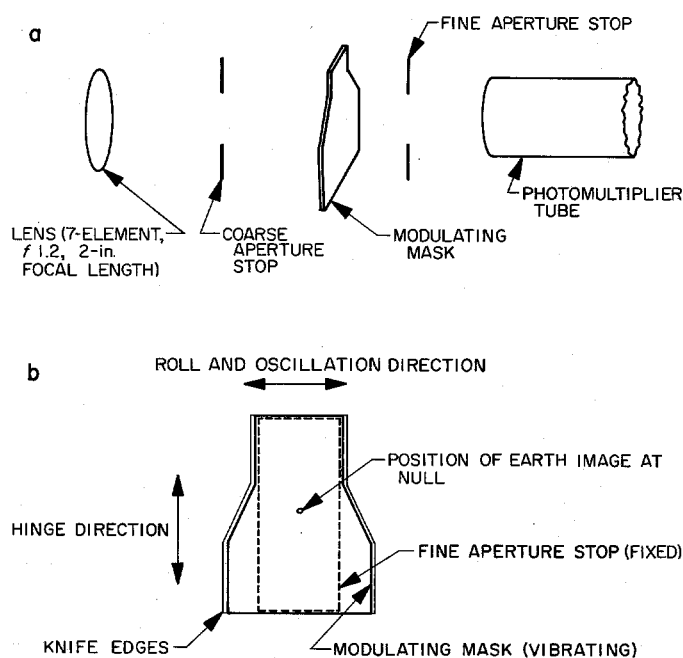


Figure 12. Relative position of LRES optical elements

tube life and has been defined as 5×10^{-7} w/cm². Intensities much higher than this will not affect the tube for short periods of time. With power off, the sensor may be pointed directly at the Sun for extended periods.

The accuracy of the LRES is ± 0.1 deg in roll and ± 0.25 deg in hinge over the range of 0 to $+100^\circ$ F. Expected inflight stability is approximately ± 0.02 deg in roll and ± 0.05 deg in hinge. Power dissipation is approximately 6.5 w and total assembled weight is 6.2 lb.

5. Seismic Motion Survey

During the last 2 years, it has been apparent that test results from high performance inertial sensors have been affected by random and periodic vibrational motion of the Earth. A program of tests has been initiated to determine the source and magnitude of the seismic disturbances and to evaluate the construction techniques presently in use by the inertial instrument industry to provide seismic isolation.

Tests to date have been made at six widely dispersed locations, including JPL. These sites embody three basic design techniques:

- (1) Large, relatively thick floor slabs separated from the building wall structures. Test piers are placed directly on these floors and the operators and test equipment are mounted on the isolated mass.

- (2) Small, relatively deep piers isolated from the building and floor. These piers carry only the inertial instrument under test and its positioning device.
- (3) Tuned spring-mass system, consisting of a large mass of concrete separated from the floor by coil springs. These piers carry only the instrument and its positioning device. Damping is supplied in various ways including dash pots, air cushions, or viscous liquids.

The sensors used to detect the seismic motions are Lerner-Griffith-type seismometers; one is used for vertical and one for horizontal motion. Their output is proportional to the velocity of motion and they are operable over a bandwidth of 0.5 to 10 cps. The output voltage is amplified, filtered with ultra-low frequency band-pass filters, and recorded on rectilinear oscillographic recorders. The filters are adjusted to give six different bands of information within the frequency bandwidth of the instruments. Each frequency band is recorded for approximately 1 min; the largest peak-to-peak excursion is measured for amplitude and frequency, and converted to g units.

It has been found that background noise or vibration is derived from air conditioning equipment, power generators, liquid pumps, sewage and water systems and other industrial equipment. Certain of these are present in the vicinity of every test site evaluated and, therefore, present a difficult problem in providing proper test environments. There are, in addition, disturbances from personnel and equipment moving in the immediate vicinity of the test piers.

The type of noise and the frequencies involved vary over a spectrum as low as 0.5 cps and above 60 cps with various amplitudes depending upon the soil material and the generating source. Typical noise levels run from 10^{-5} up to 10^{-3} g. As yet, no single isolation technique was found to give good isolation over the frequency spectrums noted or even over a 10-cps range.

D. Engineering and Mechanics

Significant accomplishments during the reporting period include the assembly of the flight spacecraft and the completion of the majority of planned developmental

tests. The spacecraft assembly operation has progressed smoothly and rapidly due, at least in part, to the many mechanical compatibility checks performed as part of the development test program. Many interface problems were resolved in this fashion before assembly of the flight spacecraft commenced. Several problem areas are currently receiving attention, but none will cause program delays.

1. Spacecraft Development

The primary effort in spacecraft development during the reporting period was concerned with changes arising from the structural qualification tests, and investigations into suspected problem areas. Aside from revising a few fastener failures, only one design change has resulted from the structural qualification tests.

a. Reduction of long range Earth sensor resonance. During the low frequency vibration tests, the simulated long range Earth sensor (LRES) assembly exhibited a resonant frequency of 25 cps with a peak acceleration of 45 g. Since this was not tolerable by the LRES, means of reducing the level were investigated.

Because of the geometric limitations, the LRES is supported from the high gain antenna hinge points by a complex yoke assembly (Fig 13).

The antenna is supported during boost between rubber pads on the spacecraft and on the adapter (Fig 14). Although the yoke itself was quite rigid, the motion of the antenna on the rubber pads, deflection of the antenna, deflection of the hinge and support structure all com-

bined to give a soft suspension. Attempts to stiffen all these areas nearly doubled the resonant frequency but did not reduce accelerations on the LRES to acceptable levels.

A pair of spring-loaded, concentric tube viscous dampers were chosen as the best compromise solution to damp motion between the LRES and the adjacent structure (Fig 15). Since the antenna with the LRES must swing away from the structure after boost, the dampers could not be attached directly to the LRES. Although the damper controls only half of every cycle (the spring returning it the other half cycle), it decreased the peak acceleration in the low frequency range to less than 1 g

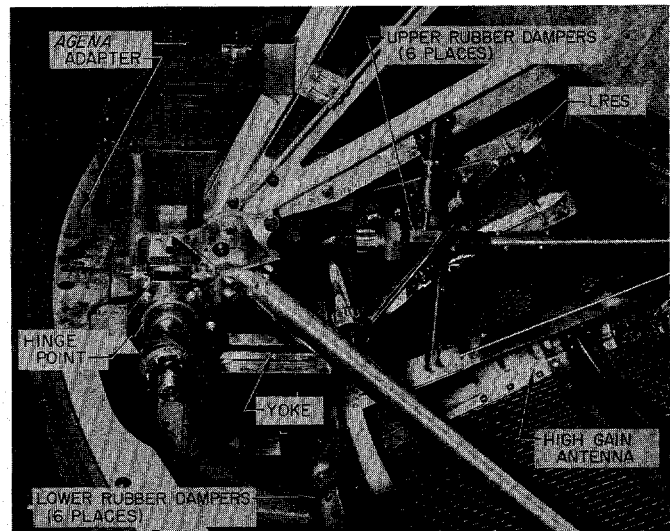


Figure 14. High gain antenna boost position

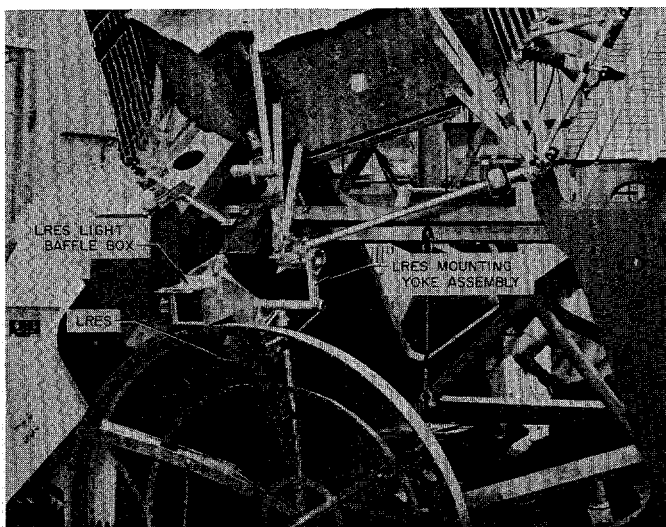


Figure 13. LRES and high gain antenna yoke assembly

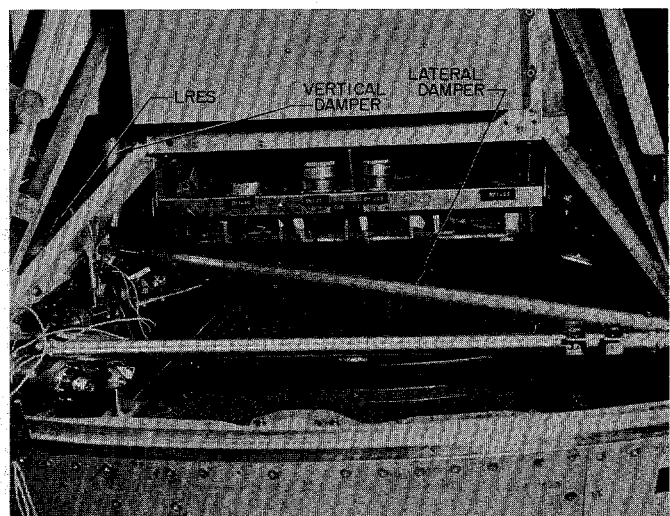


Figure 15. LRES vibration dampers installed

(a reduction factor of 50 from the original). Higher frequency resonances were damped to acceptable levels (3.5 g max). Two dampers at an angle were used to damp both axial and lateral motions of the LRES.

b. Decontamination procedures. To ensure no contamination of optical surfaces on the spacecraft (electro-optical devices and temperature control surfaces), developmental tests were made to establish AMR handling procedures. The *Ranger* approach will be used, with deviations to account for the removal of the weighty sterilization diaphragm.

Tests have demonstrated that, by continuous purging with dry nitrogen, the relative humidity inside the shroud can be kept acceptably low, even with openings of significant size in the adapter (e.g., access doors). Further tests are expected to verify that with the bottom transport diaphragm removed (just before mating to the *Agena*) a constant efflux of N_2 will minimize chances of spacecraft contamination.

Acceptable means of sealing the forward portion of the *Agena* have been developed, thus assuring no flow of *Agena* cooling air into the spacecraft compartments.

2. Structures and Dynamics

Structures and dynamics efforts during the reporting period were primarily devoted to the testing of prototype hardware and evaluation of flight hardware. Certain detailed analyses have been completed including (1) a structural and dynamic analysis of the radiometer, (2) an evaluation of the capability of the systems test stand to support the spacecraft, (3) the design and analysis for a fixture which rotates the spacecraft about the axes through the magnetometer, and (4) structural and dynamic analyses of the solar panels prepared by Ryan Aerospace Company.

Four major tests have been undertaken. Each of these tests was performed on the structural test model of the spacecraft.

a. Modal survey test. During this test, the natural modes of vibration of the spacecraft were excited by a number of small 25-lb force electromagnetic shakers. After each resonance was excited, the mode shape was measured with two variable reluctance accelerometers, one of which was moved from point to point along the structure, while the other was held stationary as a reference. Following the measurement of each mode shape, the excitation was removed, and the resulting decay curve was recorded on an oscillograph. The damping ratios for each mode were then obtained from the logarithmic decrement of this curve.

The test was performed in two parts:

- (1) The spacecraft was rigidly supported at the separation plane.
- (2) The spacecraft was supported by the *Agena B* adapter which, in turn, was rigidly supported at its base. Figure 16 shows the spacecraft during the latter phase of the test.

The test was undertaken with two major objectives in mind. The primary objective was to determine the natural frequencies, mode shapes, and damping ratios of the spacecraft which were of significant importance to design. A secondary objective was to predict, with the aid of a newly developed computer program, the response of the spacecraft to a forced vibration similar to that occurring during flight or applied to the spacecraft during type approval and flight acceptance vibration tests.

A summary of the primary modes of vibration is presented in Table 5. It is interesting to note the significant

Table 5. Summary of primary modes of vibration, boost configuration

Description of mode	Without adapter		With adapter		Change in frequency, %
	Frequency, cps	Damping ratio c/c_c (avg)	Frequency, cps	Damping ratio c/c_c (avg)	
Torsion of solar panels	—	—	22.5	—	—
Fundamental bending in y-direction	39.6	0.0150	35.8	0.0167	-9.6
Fundamental bending in x-direction	42.5	0.0125	38.5	0.0184	-9.4
Second bending in x-direction	—	—	45.6	0.0125	—
First torsion	—	—	45.8	0.0087	—
Solar panels out of phase	58.1	0.0148	55.6	0.0124	-4.3
Solar panels in phase	54.9	0.0121	61.1	0.0154	11.3
Second bending in y-direction	79.8	—	73.9	0.125	-8.0

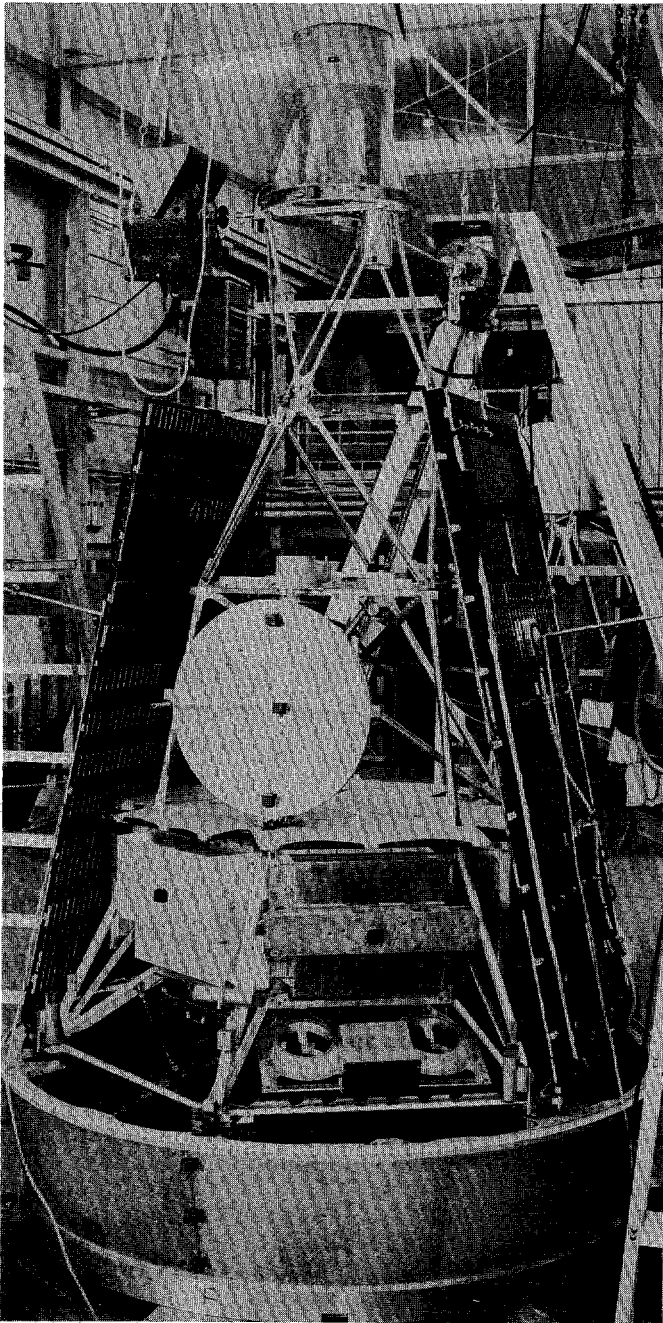


Figure 16. Modal survey of Mariner R while attached to Agena B adapter

changes in frequency which occur when the spacecraft support conditions are varied. The damping ratios, as given, are the average of ratios derived from decay curves taken at several points on the spacecraft.

Figures 17, 18, and 19 indicate the shape of several of the primary modes when the structure is supported on the adapter.

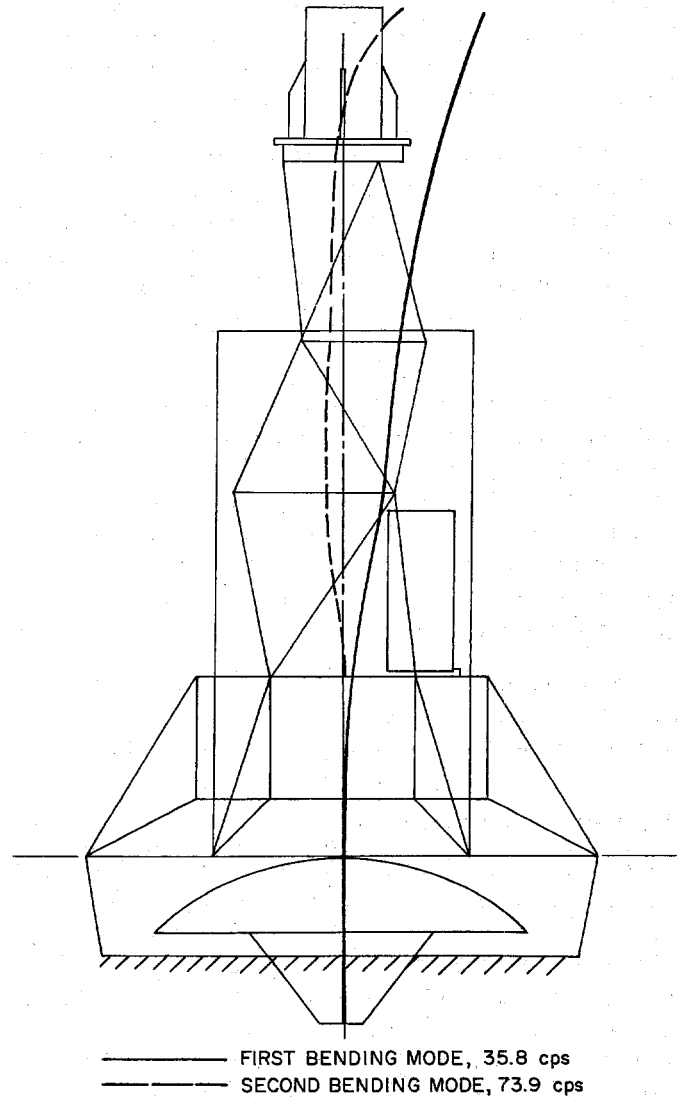


Figure 17. First and second bending modes of spacecraft in y-direction

Table 6 is a summary of the local modes of vibration which were excited during the test. An attempt was made to excite those local resonances below 100 cps which could be significant in the structural design of the components.

The prediction of the response of the spacecraft to a forced vibration has not been completed as of this reporting date. The development of this procedure will enable detailed correlation between vibration and modal survey tests, and may minimize the instrumentation required on future vibration tests.

b. Composite vibration test. This test program consisted of a series of vibration tests. During each phase of the test, the spacecraft was supported by the Agena B

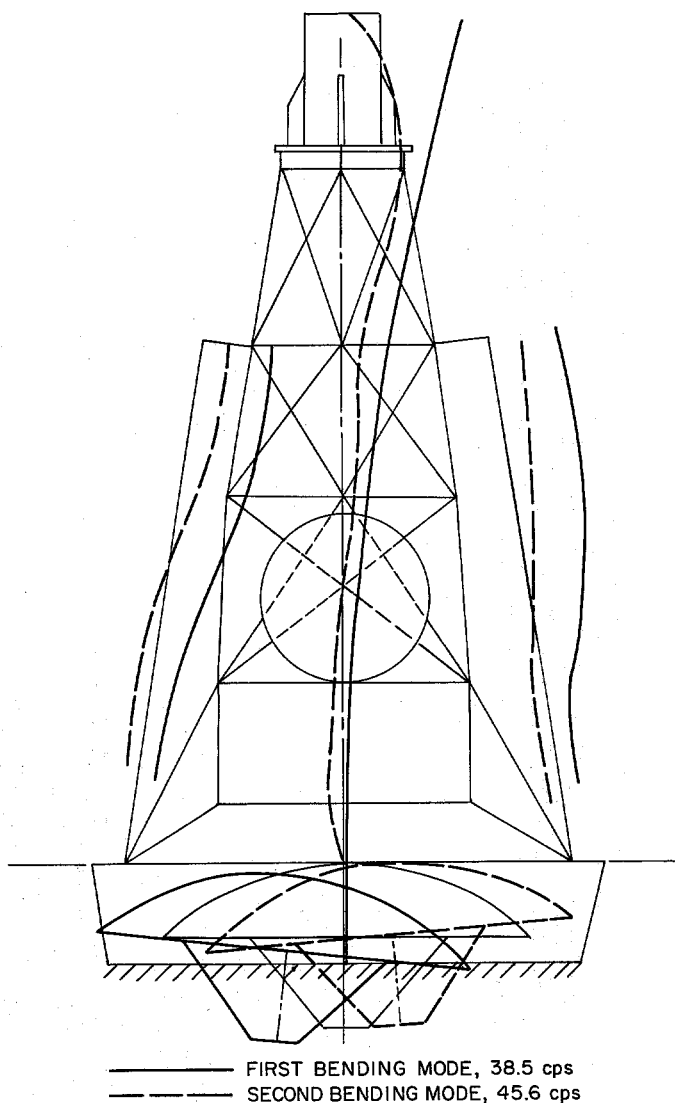


Figure 18. First and second bending modes of spacecraft in x-direction

adapter which, in turn, was supported by a fixture attached to the vibration exciter. Figure 20 indicates the spacecraft as it was supported during the x-direction, high frequency test.

The spacecraft structure successfully withstood the test loads with the following three exceptions:

- (1) The No. 4 huck rivets attaching the separation plug bracket to the tie rod became loose after the high frequency sine sweeps in the x-, y-, and z-directions. The rivets were replaced with No. 5 huck rivets and no further modifications were necessary for the remainder of the test.

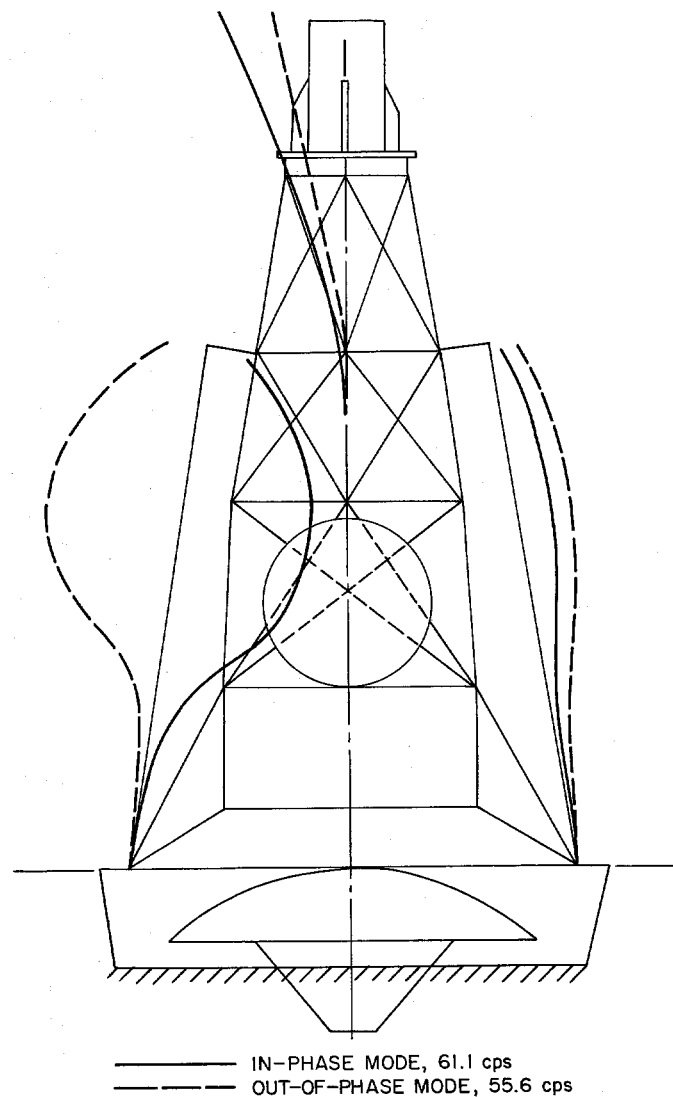


Figure 19. Mode shapes of solar panels in phase and out of phase

- (2) One of the No. 4 rivets attaching the -x solar panel latch clevis to the superstructure failed in double shear. After both rivets in this joint were replaced with No. 6 screws, the modified structure successfully passed the remainder of the test.
- (3) The high-gain antenna and LRES exhibited high gains during several parts of the test. While this structure successfully survived the test, it was felt that if electronics had been in the LRES mock-up, they would probably have experienced failure. An extensive program was conducted to provide more stiffness and damping in this area.

**Table 6. Summary of significant local modes of vibration
(Spacecraft mounted on adapter)**

Location of mode	Frequency, cps
High gain antenna and Earth sensor	26.5
Separation connector at Foot B	58.3
Attitude control jet at Foot F	67.9
Attitude control jet at Foot C	70.3
Attitude control jet at Foot E	77.1
Outer edge of upper thermal shield	77.5
Separation connector at Foot D	82.8
Midpoint of upper thermal shield	102.5

Table 7 indicates representative gains which were exhibited by the structure during the various phases of the test. This data is based on early data reduction and may be subject to change. Significant reductions in the response of the high gain antenna feed will occur due to the modifications in the high gain antenna and LRES structure.

Before and after each significant phase of the test, the spacecraft structure was subjected to a frequency test using the modal vibration system. The object was to determine any changes in the significant resonant frequencies. No large changes in frequency were observed except after the low frequency sweep in the x -direction when the solar panel latch clevis became loose. This condition showed up in the frequency check test in that one significant resonant frequency was missing after the rivet failure. After the rivets had been replaced with screws, the missing frequency response reappeared in the proper magnitude.

c. Combined vibrations test. This test was performed with the spacecraft, Agena B shroud, and Agena B adapter assembled together as a unit. The primary objective of this test was to verify clearance between the dynamic envelopes of the shroud and the spacecraft under the extreme steady-state flight vibration environment as it is presently known.

The dynamic deflections of the critical interface locations were successfully measured and found to remain well within the prescribed dynamic envelope of the spacecraft.

d. Dynamic characteristics required for autopilot stability computations. Calculations have indicated that spacecraft natural frequencies in the range of 2 to 30 cps during the mid-course propulsion mode of flight could

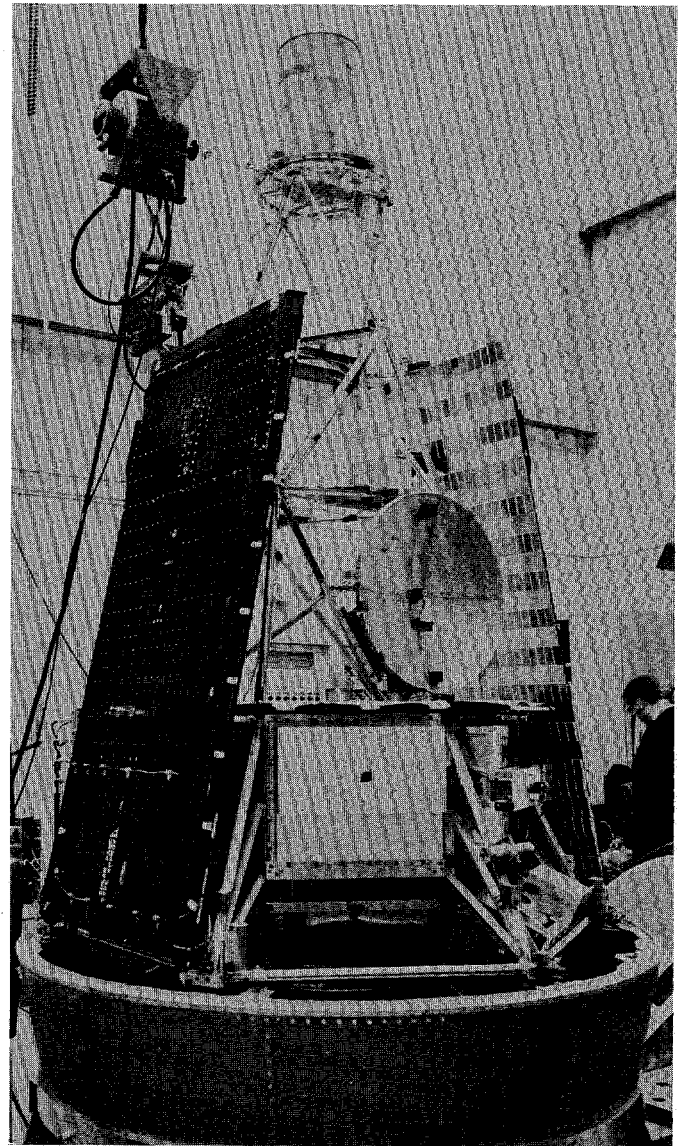


**Figure 20. Mariner R spacecraft supported for
 x -direction, high frequency test**

destabilize the autopilot, depending on the mass, moment of inertia, location, and damping characteristics of the component resonating.

To investigate this problem, a series of modal survey tests were performed on the spacecraft. The mid-course configuration was simulated by placing the spacecraft on the systems test stand, extending the solar panels and high gain antenna, and unlatching the radiometer as necessary. Figure 21 gives an indication of this test setup.

The fundamental frequency of the extended solar panel and the fundamental torsion frequency of the extended

Table 7. Gains exhibited during composite vibration test

Accelerometer location	Direction of excitation					
	x		y		z	
	Frequency, cps	Q	Frequency, cps	Q	Frequency, cps	Q
Top of omni-antenna (Direction of shake)	39	30	36	38	32	3.5
	56	60	85	9.0	95	5.0
Tip of solar panel (+x) (x-direction)	31	33.3	25	21.5	—	—
	56	28	31	14.5	—	—
Center of solar panel (+x) (x-direction)	31	18	31	5.4	—	—
	85	20	85	4.4	—	—
Outer edge of radiometer (-x) (y-direction)	39	24	39	6.4	—	—
	56	14	69	12	—	—
Chassis No. (Assembly 6) (Direction of shake)	31	4	—	—	—	—
	66	3.2	—	—	—	—
High gain antenna feed (x-direction)	13	9.3	36	3	—	—
	36	10	69	1.5	—	—
High gain antenna feed (x-direction)	39	9.6	36	6.5	—	—
	56	18	45	8.7	—	—



Figure 21. Modal survey test setup for autopilot stability analyses

high gain antenna were found to fall within the critical range. In order to eliminate these conditions as problem areas, the following modifications are being made to the spacecraft:

- (1) The solar panel frequency has been reduced from 6.5 to 2 cps by replacing the aft monoball joint on the solar panel actuator with a cylindrical rubber bushing.
- (2) The damping ratio in the high gain antenna was increased from $c/c_c = 0.0179$ to $c/c_c = 0.0835$ during the torsion mode through the installation of a concentric tube grease-in-shear damper.

The natural frequencies in or near this critical range, together with their respective damping ratios, are presented in Table 8. The spacecraft, with the modified dynamic characteristics, will have no tendency to destabilize the autopilot.

3. Mechanisms

a. Antenna yoke assembly. Figure 22 shows the high gain antenna yoke with the LRES mount, and baffle box. As stated in SPS 37-12, it was necessary to mount the sensor at an angle (approximately 30 deg from vertical) and to use a mirror so that the sensor could see in the antenna pointing direction.

It was later determined that the Earth sensor is sufficiently sensitive to diffuse light from sources outside of its normal field of view. This made it necessary to

Table 8. Summary of primary modes of vibration, mid-course configuration

Component, mode	Frequency, cps	Damping ratio c/c_c
Solar panel, about hinge axis	2.00	0.0275
High gain antenna, about actuator axis	2.66	0.055
High gain antenna, combined bending and torsion	3.15	0.0624
High gain antenna, torsion about pivot arm	5.4	0.0835
Radiometer, torsion about hinge axis	6.2	0.35
Solar panel, torsion	14.2	0.0398
High gain antenna, about actuator axis	15.9	0.1005
High gain antenna, breathing	27.3	0.047
Solar panel, bending in plane of panel	32.0	0.0317
Solar panel, about hinge axis	43.5	0.0106

block reflections from the antenna feed and rim and to provide more Sun shielding during the period when the Sun-probe-Earth angle is near the minimum value. To block this light so that it could not get to the mirror, a baffle box was designed so that the elements of the box are out of the sensor's field of view and stray light coming in at oblique angles is absorbed.

To achieve a black, knife edge on each baffle, a unique method of construction was used. Thin (0.002 in.) strips of black, oxidized brass are inserted through slits in the box, aligned and bonded in place. The edges of the strips are rounded, thus providing an edge of approximately 0.001-in. radius. The protruding outside edges are then trimmed and aluminum foil is glued over them to assure a light-tight seal and to present a low emissivity surface to space. Preliminary tests indicate that the baffle box is effective, but more extensive tests are planned to assure solution of the problem.

b. Radiometer scan actuator. The function and general operation of the actuator was described in SPS 37-12. Figure 23 shows a flight actuator with its cover removed. The output end of the actuator is in the foreground and the two drive motors are at the rear. The uppermost motor-gearhead in Figure 23 is the fast scan drive which drives one side of the differential directly. The other motor drives the near side of the differential through a 10 to 1 gear reduction. The differential carrier in turn drives the output through two more stages of reduction with a slip clutch between the two. The function of the clutch is to protect the differential and reduction gears from shock loads during boost or from overloads by improper handling of the radiometer during systems operations

Two travel stop switches reverse the scan. The arm which operates the switches has a cam and spring drive

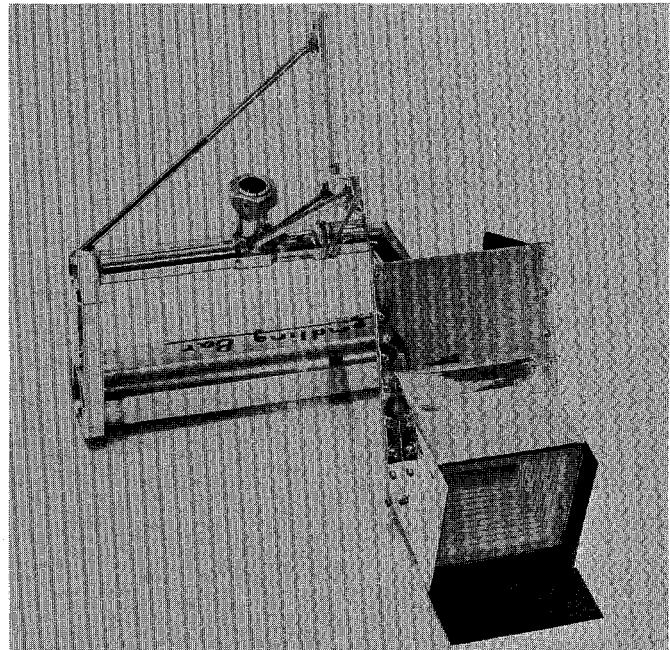


Figure 22. Antenna yoke assembly with LRES mount and baffle box

to take care of overtravel to avoid damage to the switches. The potentiometer, directly connected to the output, provides feedback of actuator position.

A prototype model of the unit has been mounted on the structural test spacecraft through the type-approval (TA) tests of that unit. The actuator functioned normally at the conclusion of several TA shakes. Included in the plans for TA testing of a flight type actuator is a 75-hr run in a vacuum chamber with the cover completely vented. This represents a much more severe test than the anticipated flight conditions where the run is about one-third as long. Some preliminary tests have been run in a vacuum chamber with partial units to determine the temperature range in which the motors will operate. These tests indicate that the temperature control requirements are not severe.

The units delivered for flight have been vibration tested to flight acceptance levels following an 8-hr run-in. The purpose of the run-in is to burnish the MoS₂ coating, used as a lubricant on the gear teeth, and reveal any defective or contaminated ball bearings in the motors or gear heads. Thus far, the run-in procedure has proven of value for discovering a dirty bearing in one drive motor and one defective switch. At the conclusion of the test, the box is pressurized to 2 atm with filtered dry air and is then leak tested.

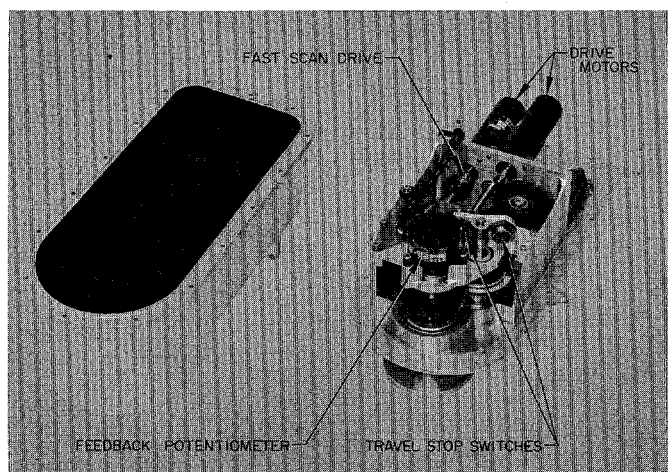


Figure 23. Radiometer scan actuator with cover removed

4. Materials

a. Stable blacks. Three 100-hr exposure tests have been concluded in which seven black surfaces were irradiated with ultraviolet. The irradiation is assumed to be equal to six solar intensities; therefore, the test run approximated 75 days in the Sun. Tested were three black pigmented paints, a hard black anodize, several black oxides, and a Dow 7 treatment.

The black pigmented paints exhibited a slight change in solar absorptance, while the emittance at 300°K was not changed. All three paints (*Sergeant* black, *Laminar* X-500 black, and *Cat-A-Lac* black) can, therefore, generally be considered suitable for *Mariner* (or *Ranger*) surfaces which face the Sun.

Black oxides such as black copper oxide and black nickel appeared to survive the exposure with no apparent change in the original reflectivity.

Hard black anodized 6061-T6 similarly showed no change in reflectance after the 300-hr exposure and, along with the two black oxides, can be assumed to be stable for at least several months.

A Dow 7 treated magnesium sample, checked before and after exposure, exhibited an increased reflectance after the exposure. The change existed in both the solar absorptance region and the range of wavelength comprising emittance at 300°K. However, the resulting change was much more evident in the infrared region; thus, the change affected the emittance to a larger degree. Because this was the first test of Dow 7 in ultraviolet, more tests will be run to verify the results.

b. Stable whites. Extended ultraviolet testing has defined the white surfaces with minimum yellowing (change in solar absorptance) for use on *Mariner* R and provided data on approximate solar absorptance that can be anticipated at the termination of the Venus mission. Additional tests are being conducted to check the effects of various lots of materials, handling, cleaning, etc.

Measurements of several samples indicated no change in emittance due to ultraviolet and vacuum. Experience in this program indicates caution must be exercised in using ultraviolet degradation data from tests conducted in air. Results of tests where vacuum leaks occurred show $\frac{1}{2}$ to $\frac{1}{4}$ the degradation obtained with high vacuum.

E. Propulsion

The function of the *Mariner* R mid-course propulsion system is to remove or reduce *Agna* B injection dispersion errors so that a Venus flyby with a sufficiently small miss distance can be reasonably assured. This function is performed during the single spacecraft mid-course maneuver executed approximately 8 days after launch. At that time the spacecraft is directed to turn to a prescribed attitude in space and impart a corrective impulse via the mid-course propulsion system. A description of the *Mariner* R mid-course propulsion system was presented in SPS 37-12 and -13.

Since the last reporting period, the *Mariner* MR-1 and -2 flight systems have been delivered to the spacecraft assembly facility (SAF) and installed in the flight spacecraft. The MR-3 spare flight propulsion system buildup is essentially complete and will be delivered to the SAF as a spare assembly.

During the course of engine qualification testing, utilizing the new dual bridgewire squibs in the engine start and shutoff explosive valves, local yielding and, in some cases, rupturing of the normally closed explosive valve trigger assembly was observed. In the cases where rupturing occurred, the normally closed valves failed to operate. These failures can be attributed to the differ-

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ent pressure-time characteristic of the dual-bridgewire squib compared to the single bridgewire squib originally designed for the system. It should be mentioned, however, that the trigger assembly would probably not have ruptured had the assembly been torqued properly at installation. With correct torquing the units appear only

to yield. In order to alleviate the problem, it has been decided to thicken the walls of the trigger assembly to better withstand the overpressure condition and ensure successful valve actuation. The new trigger assemblies will be tested and retrofitted to the MR-1, -2, and -3 propulsion systems as they become available.

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III. Mariner B Project

The primary purpose of the *Mariner B* project is to permit scientific investigations of the planets Venus and Mars during their periods of availability in 1964 through 1967. Secondary purposes in order of priority are: (1) to make interplanetary scientific investigations in the regions between Earth and Mars, (2) to develop experience in the design of a spacecraft that has considerable flexibility with regard to the specific missions for which it is basically capable of covering, (3) to provide experience and knowledge which will permit a quality and efficient design of the later *Voyager* spacecraft. To these ends, the spacecraft will be designed for a precision flyby mission and will incorporate the capability of either carrying or not carrying a small landing capsule.

The *Mariner B* spacecraft will be injected by the *Centaur* launch vehicle. Two identical probes are expected to be launched at each of the planet launching opportunities. Deep space probes, scheduled between periods of Mars and Venus availability, are also included in the *Mariner B* project.

Mission objectives and design criteria were generated and have been used for the preliminary design effort. This effort was initiated in October 1961, is proceeding

on schedule, and will be completed on April 1, 1962. Design characteristics and restraints have been completed and will be used to generate the functional specifications scheduled for completion on April 1. A prerelease Project Development Plan (PDP) has been completed. Copies of this plan were transmitted to NASA Headquarters and Marshall Space Flight Center for coordination during the week of February 26. A final coordinated issue of this PDP is scheduled for release on April 1.

A. Guidance and Control

1. Computer Controller

An improved 80-phase clock has been constructed for the computer controller which has reduced the number of transistors from 18 to 4. Current steering commutating switches are used replacing the anti-coincidence switches resulting in a much greater signal-to-noise ratio on the output lines. Start-stop circuitry has been developed in conjunction with the 80-phase clock to allow single word

time operation of the computer, which is a necessity for efficient controller operation. The clock has been tested to 348 kc, some 4 to 8 times the contemplated operating frequency. The dc to pulse power conversion efficiency at 80 kc is 42% at a power input of 4 w. The power input is a linear function of frequency decreasing to less than 100 μ w at near zero frequency.

An improved dual-transistor input stage for the transfer circuits has been designed utilizing a newly developed input transformer core. This circuitry redesign has resulted in the elimination of seven components in each transfer circuit. The redesign has also greatly simplified the strobing circuitry which allows the transfer circuit to respond only to the *read* outputs of the logic and register cores.

Work is continuing on low semiconductor count memory access switching circuitry. A unipolar pulse output circuit, which may be driven in an anticoincidence mode, has been devised. This device utilizes a square loop core and a transformer core in combination to produce unidirectional current pulses suitable for use with linear-select memories. The diode normally associated with each word in a word select memory can be eliminated by the use of this circuit, thus making it attractive for reliable memory systems.

A study of the methods of operating the logic cores in inhibit core mechanizations has led to the result that the cores may be switched in a pure inhibit current mode such as that used in a *rope memory*. This mode of operation lifts the restrictions imposed on the core characteristics by coincident-current methods and allows lower coercive-force, higher-output cores to be used. Current compensation as a function of temperature is not required in this mode of operation. Another benefit obtained is that larger diameter cores may be used, which will simplify the winding of the logic inhibit wires.

2. Planet Tracker Precision Actuator

The *Mariner B* approach guidance maneuver requires the knowledge of the magnitude of angles between the target planet and two other celestial references. It is contemplated that these angles will be measured by using an actuator to move the optical axis of a sensor until a null is reached, and then measuring the position of the actuator output shaft.

Two techniques are available to accomplish this measurement: (1) An angle measuring device with sufficient accuracy and resolution could be attached directly to the actuator output shaft. (2) A transducer, that does

not have sufficient accuracy and resolution at its input shaft, could be attached to the actuator output shaft by means of a precision mechanical transmission. An investigation showed that the second technique was the least complex of the two.

It was, therefore, decided to use a digital shaft encoder connected to the actuator output shaft through a precision gear train. This combination gives the desired resolution and has the capability of meeting the estimated accuracy requirements. The advantage is also gained that the data is available in a form convenient for telemetry purposes. A contract has been let to the United Shoe Machinery Corporation for the development of a precision actuator employing the harmonic drive. The harmonic drive (Fig 1) is composed of a wave generator, a flexspline, and a circular spline. The wave generator (which can be thought of as the inner race of an elliptical bearing) is the input. The flexspline forms the outer race and has teeth cut on its periphery. The circular spline is an internal gear having two more teeth than the flexspline. Thus, depending upon which member is fixed, each time the wave generator turns through one revolution, either the flexspline or circular spline moves an amount equivalent to two teeth.

Inaccuracies in a precision spur gear train come from several sources: tooth placement error, tooth shape error, eccentricities, and gear train compliance with torsional load. Because of the many tooth contact points in the harmonic drive, tooth placement and shape errors tend

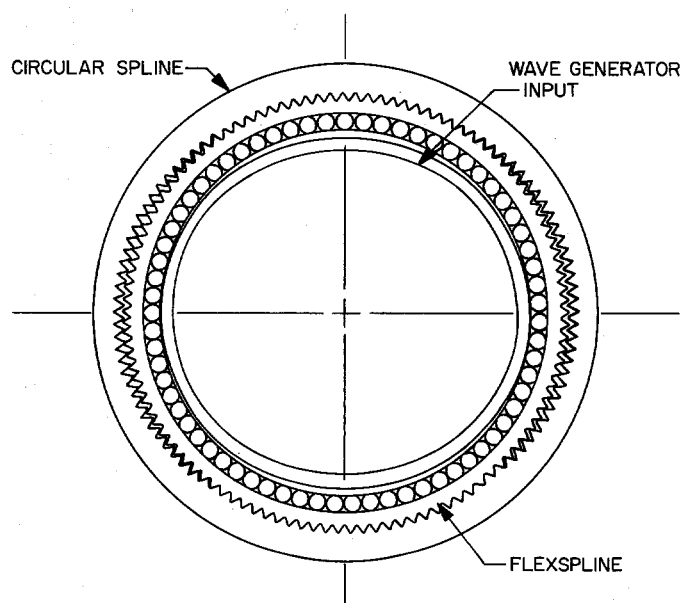


Figure 1. Harmonic drive

to be averaged, and thus their effect is minimized. Also, fixed centers are not required, which eliminates errors due to eccentricities. Other advantages of the harmonic drive are its high torsional rigidity, minimum backlash, and the fact that lapping can be used to increase accuracy.

3. Antenna System

The narrow beamwidth, high gain antenna is part of the deep space telemetry link between the Earth and the spacecraft. Since the *Mariner B* spacecraft is not Earth-oriented, the high gain antenna requires 2 deg of freedom and control. Each axis is controlled by a program generated in the central computer and sequencer (CC&S). The antenna angle program starts from an initial preset angle and continues as an incremental function. The antenna erects to the preset angle on receipt of a command from the CC&S. In this mode the antenna operates as a closed loop digital servo as shown in Figure 2. The output of a digital shaft position encoder measuring the antenna angle is compared with the preset angle which is stored in digital form in the antenna control system. The digital error signal drives the antenna actuator until the preset angle is reached. Then the control system is switched from the closed loop mode to an incremental mode in which the actuator is controlled directly by the CC&S program. The CC&S controls the antenna by transmitting a pulse for every incremental motion of the antenna. Simultaneous ground command is possible by adding command decoder pulses to the

CC&S program output. The preset angle stored in the command register can be cleared and reinserted by the command decoder. Upon receipt of a command the system can be returned to the closed loop mode, and thus be oriented to any angle which has been inserted into the register.

The actuator contains a stepper motor as prime mover, a gear train to increase its resolution and torque, and a digital shaft encoder. The actuator is designed so that the output shaft is capable of driving a 10 ft-lb torque load at an angular velocity of 0.2 deg/sec. The torque load on the actuator is composed of: the friction of the various bearings, seals, and rotary joints; the torques required to bend antenna cabling; and the inertia loads caused by accelerations or decelerations of the antenna.

The digital comparator is mechanized so that the reference and feedback signals are compared only at an interval of time which is greater than the minimum time between steps of the stepper motor. When a comparison is made and the difference is one bit or greater, the comparator emits a pulse. This pulse is accepted by the motor drive electronics and held while it drives the stepper motor through a predetermined number of increments. At the end of this interval the comparator may or may not emit another pulse depending upon the magnitude of the reference signal and the new feedback signal.

A block diagram for use in the analysis of the control system is shown in Figure 3. In the representation of the digital comparator, the output of a summing junction goes to an on-off switch with dead-band. Whenever the reference signal and the output of the digital shaft encoder differ by less than the least significant digit, the output of the on-off switch is zero. If the difference is one bit or greater, the output of the switch is either $+E_c$ or $-E_c$ depending upon the polarity of the difference. The switch output at the rate at which the pulses are emitted from the actual comparators is sampled by Sampler S1. The output of S1 is thus either zero or a series of pulses. The motor drive electronics is represented here as a zero order holding device. The function of this block is to hold the output of the comparator during the interval when S1 is not sampling, and thus simulate the pulse train that is initiated by the comparator pulse. Even though the output of this block is not a series of pulses, the simulation achieves the same effect by proper representation of the stepper motor.

In this representation the output of the zero-order hold is integrated at the rate $E_c K$ v/sec. This produces a ramp function which is sampled by Sampler S2. The

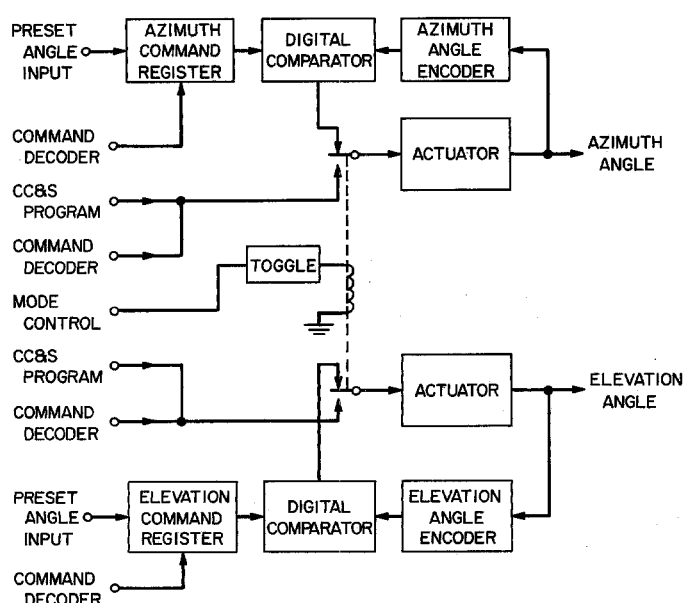


Figure 2. Mariner B antenna control system

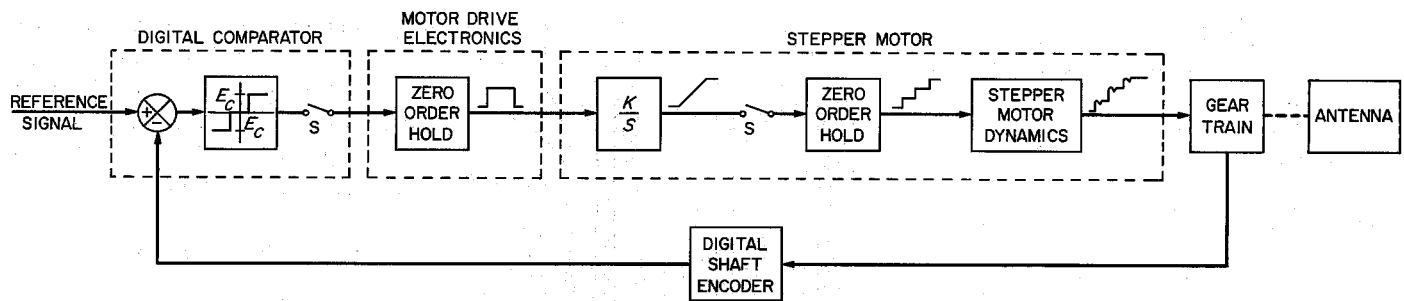


Figure 3. Simulation block diagram for single axis of antenna control system

output of the holding block following S2 is then a staircase function. The stepper motor dynamics, which can be approximated by second-order transfer function, modify this staircase to give the motion of the stepper motor output. The loop is closed through the actuator gear train and the digital shaft encoder. This block diagram (Fig 3), which is the first approximation of the analytical representation of the system, is to be used for predicting the stability and tracking performance of the antenna servo.

The digital servo logic problem is a number comparison problem. The number representing the output angle is compared with a number representing the required output in order to extract the position error. This comparison must remain valid at all possible transition values of the output number, since the output takes on a continuous range of values. In binary codes where many digits change simultaneously, difficulties occur because these changes are not exactly simultaneous. This produces an error in the comparison during transition from one output number to another.

To avoid this transition error on ambiguity problem, two alternate approaches are possible. One is the use of a unit distance code such as the gray code in place of the binary code. The unit-distance codes have the property that only one digit changes on transitions between consecutive numbers. Thus, the problem on nonsimultaneous changing is completely avoided. A second approach is to use a binary code encoder with multiple contacts and/or tracks controlled by external logic circuitry. In such a scheme the transitions of all the higher order bits are controlled by the least significant bit. Encoders utilizing such techniques are referred to as nonambiguous binary encoders.

Both gray code and nonambiguous binary encoders are being evaluated for use in the antenna system. The gray code has the advantage of simplicity of the encoder but makes the comparison difficult. The nonambiguous

binary encoder allows the use of a greatly simplified comparator.

The comparator compares the encoder output with the number stored in the command register and determines the sign of the difference. In a system where the magnitude of the difference is not required, a full subtraction of the two numbers is unnecessary. Subtractions are undesirable in digital servos because carries propagate from the least significant digits. Where large numbers are involved this propagation time limits the speed of the comparator which, in turn, limits the speed of the servo. This sign of the difference can be obtained from a bit-by-bit comparison of the two-number; the sign of the difference is the sign of the most significant inequality. The logic diagram of such a comparator is shown in Figure 4 where

$$A = \sum_{n=0}^{n=k} a_n 2^n$$

$$B = \sum_{n=0}^{n=k} b_n 2^n$$

are the binary numbers being compared. In comparing two digits a_n and b_n , it is seen that $a_n > b_n$ only if a_n is a "1" and b_n is a "0." Similarly, $b_n > a_n$ only if a_n is a "0" and b_n is a "1." The Boolean expressions for these requirements are

$$a_n > b_n \quad \text{if } a_n \bar{b}_n = 1$$

$$a_n < b_n \quad \text{if } \bar{a}_n b_n = 1$$

where

$$\bar{b}_n = 1 \quad \text{if } b_n = 0$$

$$\bar{b}_n = 0 \quad \text{if } b_n = 1$$

The inversions indicated by $\bar{a}_n$ and $\bar{b}_n$ are performed by the block marked "I." The a_n and b_n digits are combined in an *and* gate, the output of which is a "1" only if both inputs are "1." In Figure 2 the a_n , b_n comparison has separate outputs for $a_n \bar{b}_n$ and $\bar{a}_n b_n$. It is seen that at any digit comparison an output of either $a_n \bar{b}_n$ or $\bar{a}_n b_n$ will

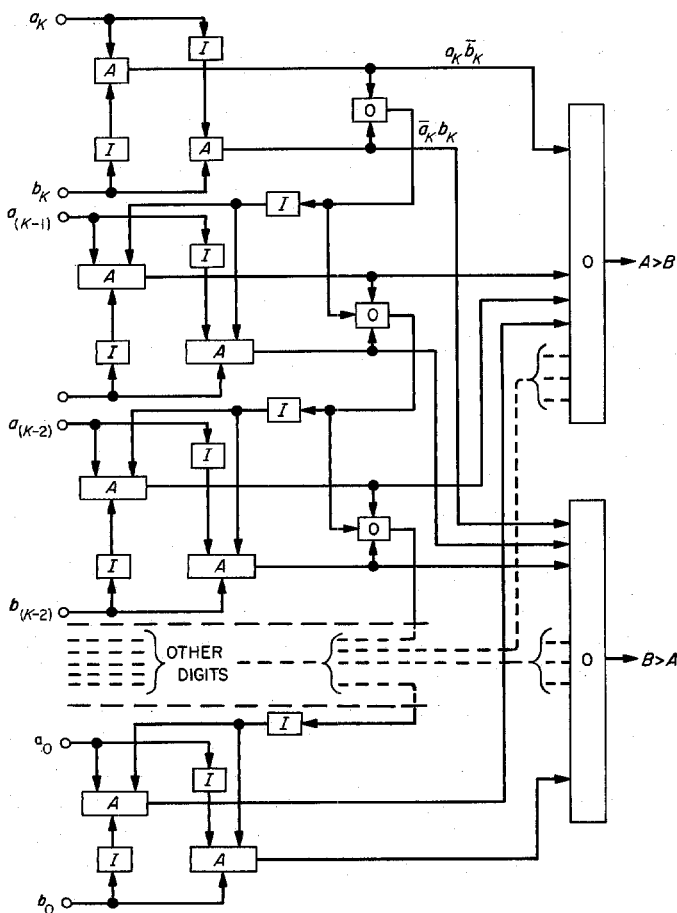


Figure 4. Logic for a sign-only binary comparator

route a "0" to one input of each *and* gate of all lower order comparisons, thus inhibiting their operation. Thus the output of the over-all comparator is the output of the most significant nonzero comparison.

This type of comparator is called a parallel comparator since parallel comparisons are made of all K -digits. A second type would be a serial comparator in which the K -digits are compared sequentially starting with the most significant digits. Such a comparator has the advantage that the total number of logic building blocks is greatly reduced. However, it does require the use of either a solid state commutator or of a ring counter for each input number. At present both types of comparators are being considered for the *Mariner B* application.

B. Spacecraft Design

With the preliminary design of *Mariner B* ending, attention is being directed to specific problem areas and

to generating functional specifications. Design layouts are being generated in an effort to fully understand all of the major interface problems. In the case of such long lead-time components as the fold-out solar panels, fairly detailed design is being pursued to minimize the effort required when hard design commences.

1. General Configuration and Mechanisms

General configuration of *Mariner B* remains essentially the same as reported in SPS 37-13, except for detailed modifications in the packaging area and some fairly minor dimensional changes. Figure 5 shows three views of a $\frac{1}{10}$ -scale model of the spacecraft.

Dimensional changes were required as a result of a more definitive envelope and interface specification with the *Centaur*. The launch vehicle dynamic envelope has been jointly defined by JPL and General Dynamics Astronautics (GDA).

The total solar panel area was increased from 130 to 145 ft² in order to meet power requirements. This required lengthening of the fold-out solar panel to about 78 in. Also, the over-all length of the main hex was increased to allow for a more flexible cabling arrangement.

An over-all dimensional control drawing, defining all major attach points, hinge axes, and spacecraft envelope dimensions is being prepared. This, along with the over-all configuration drawing and the structure functional specifications, will complete the configuration preliminary design.

Table 1 is a preliminary estimate of the spacecraft center-of-gravity locations and moments-of-inertia for several flight modes.

2. Structural Design

The basic spaceframe structurally coordinates all of the subsystems forming the spacecraft. It has the shape of a hexagonal prism and is made up of six columns, three rings, and shear plates which serve as part of the electronic packaging chassis. The capsule is supported directly on the columns while other major components such as solar panels, PHP, and antenna are supported by discrete latch structures.

The preliminary design period has brought some detail changes to the spacecraft configuration which have resulted in detail changes in the structural configuration of the vehicle.

Negotiation with GDA has resulted in changes to the dynamic envelope limitations which permit lowering the

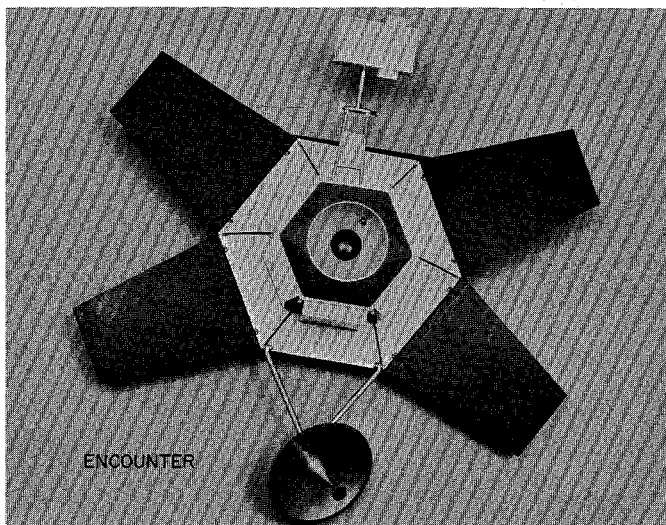
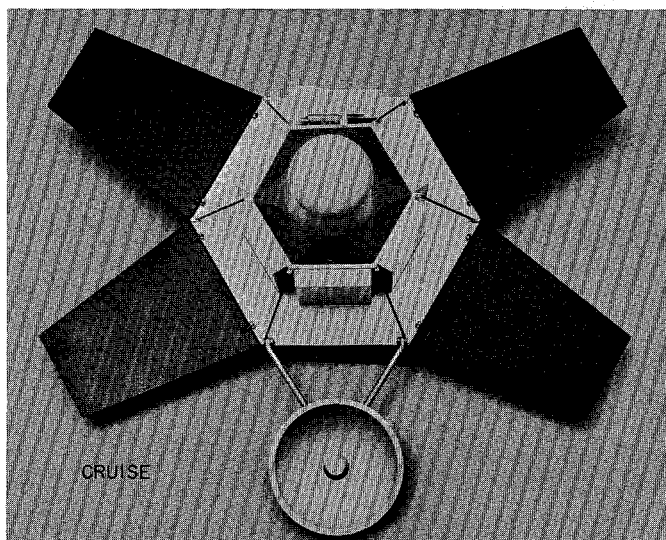
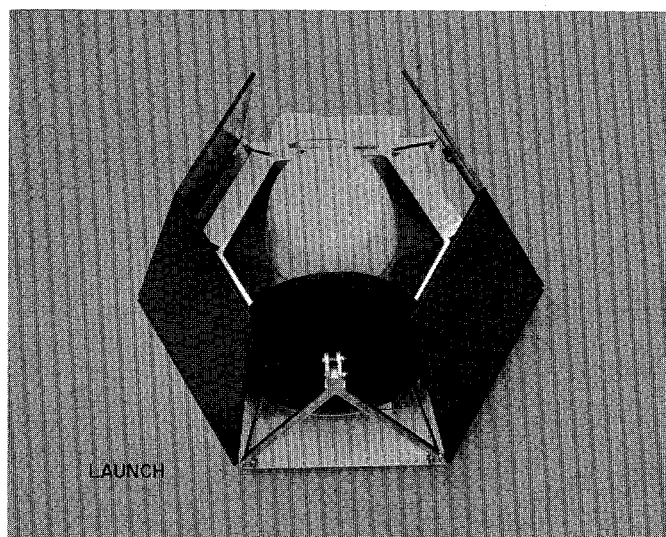


Figure 5. Configurations

Table 1. Preliminary estimates of Mariner B center-of-gravity locations and moments-of-inertia

Configuration	Wt, lb	x , in.	y , in.	z , in.	I_x , slug-ft ²	I_y , slug-ft ²	I_z , slug-ft ²
Boost with capsule	1500	0.28	-1.06	472.7	286	313	267
Cruise with capsule	1500	0.28	-1.98	476.4	365	465	521
Cruise without capsule	1250	0.33	-2.38	483.3	249	347	513

spacecraft 7 in. in the shroud. The spacecraft diameter has been increased to approximately 108 in. and the fixed solar panel gross area has increased from 48 to 54 ft².

A system requirement for ground checkout cabling versatility was met by revising the spaceframe structure to accommodate the plugs and the system cable trough inside the spaceframe at mid-height. Electronic packaging chassis took on a new geometry with double rows of boxes, one above and one below the row of plugs.

To accommodate these plugs, the spacecraft was lengthened by 4½ in., which in turn required a lengthening of the antenna boom to prevent interference. This added boom length helps offset the lengthened solar panels and minimizes the antenna beam interference.

The fold-out panel design capitalizes on experience gained through the Mariner A and R panel development programs. Similar construction (i.e., skin-corrugation) is employed, but the corrugations span transversely over two support beams which are torsionally connected at the top of the panel (Fig 6).

The four fold-out panels are to be structurally identical for maximum interchangeability and minimum spares. To facilitate ease of handling and cell layup, the panel surface is divided into three modules. Only two modules per panel require cells to meet the Venus mission power requirements. The fold-out panels are to be utilized in locating scientific experiments at maximum distances from the body of the spacecraft. Three panels will have scientific instruments mounted on the tips, including a fixed boom on one, about 2 ft in length, for accommodating a magnetometer. All panels are stiffened to the requirements of the one panel having the heaviest equipment at its tip, and a standard bolt pattern will be supplied to maintain the interchangeability characteristic.

The fixed panels are of similar construction to the fold-outs, but the support beam geometry is different (Fig 7). On the Mars mission fixed panels mount cells

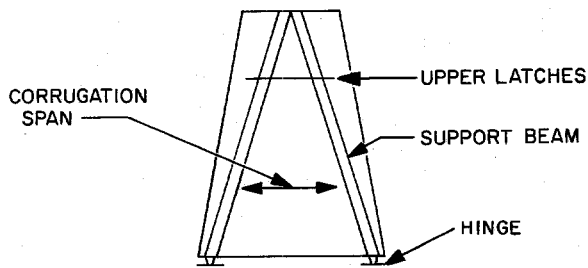


Figure 6. Mariner B fold-out panel design

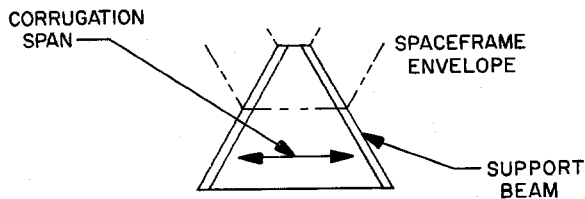


Figure 7. Fixed panel support beam geometry

and insulation required to isolate the spacecraft from the solar panels. The ability to locate holes in these panels below the packaging envelope is required for the guidance and scientific equipment which are Sun-oriented, and perhaps for temperature control of the PHP.

Design of the solar panel structures has progressed into somewhat further detail than other structures on the vehicle because of scheduling requirements for cell layup. To verify the basic corrugation section, a structural test module is being designed and fabricated for testing in early April 1962. This will permit early evaluation of the design and the earliest possible delivery of flight panels.

The high gain antenna structure is being studied to achieve the lightest reflector that can withstand the dynamic environment. In addition to a rib-wire mesh reflector similar to the *Mariner A* design, solid shell construction of aluminum honeycomb and chemically milled sheet is being considered. It is planned to conduct vibration tests on one or more types of reflectors in this evaluation.

The antenna support scheme is complicated by the 2 deg of motion characteristic to this antenna installation, and by the desire to use a thin shell reflector which must be supported at its hub. This requires that the feed be supported in a manner which prevents rotation about the two axes of motion. Latch structure from the spaceframe will attach to the boom below the antenna, completing the structural tiedown for the boost con-

figuration. Again, the antenna boom is being utilized as a means for getting certain science experiments out away from the main spacecraft structure to achieve required look angle geometry. The PHP has been selected for mounting planet scanning equipment. The scientific equipment will be arranged to meet look angles and envelope requirements, and structural members from the spaceframe will be supplied to support the hinge and tiedowns.

Handling of the spacecraft and operations sequence has received some attention. The large fixed panel array presents an access problem. The vehicle must be capable of assembly upside down since the lights in the space simulator are in the ceiling.

Easy access to the mid-course propulsion unit is mandatory. It appears that removal of all, or at least the center portion of, the fixed panels is the best method for accomplishing this. Details of the assembly process will bear heavily on the final technique selected.

By utilizing a *Mariner A* fixture, a reinforced mock-up of the spacecraft was vibrated on the 25,000-lb electronic shaker in the environmental facility. Structural type-approval-level sine sweeps were made in a transverse direction. Both shaker and the fixture supported by an oil table performed satisfactorily.

3. Structural Dynamics

A preliminary structural configuration of the *Mariner B* basic bus was fabricated and ballasted with quasi-rigid masses simulating bus-mounted equipment, mid-course rocket motor, and a capsule. Additionally, simulated fold-out solar panels having nominally representative mass distribution, but purposefully increased rigidity distributions, were constructed with two alternate means of structural attachment.

Modal vibration analyses of this simulated spacecraft, with each type of panel support, were performed for both three-point and six-point attachments to a quasi-rigid, fixed, *Centaur-Mariner* adapter. These analyses used measured mass data and derived structural data in a digital computer program initially conceived for application to pin-jointed spaceframes.

Concurrently with the performance of the modal calculations, modal vibration surveys were made of each configuration (Figs 8 and 9). These surveys entailed the use of a multichannel shaker system with test techniques appropriate to excitation of: normal modes, yielded modal

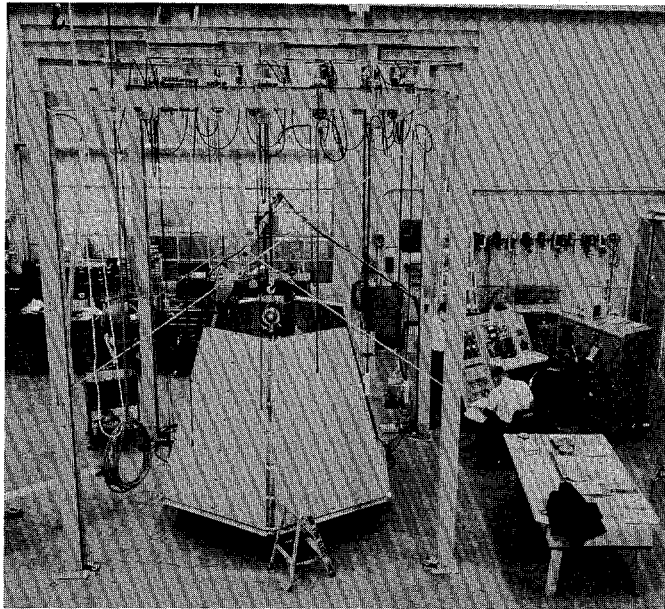


Figure 8. Modal vibration test setup, Mariner B analytical model with fold-out solar panels

characteristics defined by frequency, mode shape, and structural damping coefficients. This test data, in conjunction with the known mass distribution, are currently undergoing numerical processing to determine the quality of the testing, as judged by the degree of orthogonality of measured modes. Results, thus far, from two of five tested configurations (one without fold-out solar panels), indicate that the over-all test accuracy has been good. Accordingly, the modal characteristics derived by numerical analysis will be compared with the measured modal characteristics in a basic assessment of the chosen analytical approach to normal-mode calculation.

4. Temperature Control

The temperature control effort has been divided into three work areas: the bus, the planetary horizontal platform (PHP), and the capsule. Providing adequate temperature control of the integrated guidance package, especially the gyros, is the top priority temperature control effort. This top priority rating is due to the difficulty of holding the tight temperature limits specified and the importance of meeting them.

a. Bus. The major portions of this reporting period were spent in making analytical studies of the bus structure during maneuvers. A mathematical model of one bay of the hex was constructed and run on an analog computer. The inside face of this bay was allowed to radiate to a large cavity held at a fixed temperature.

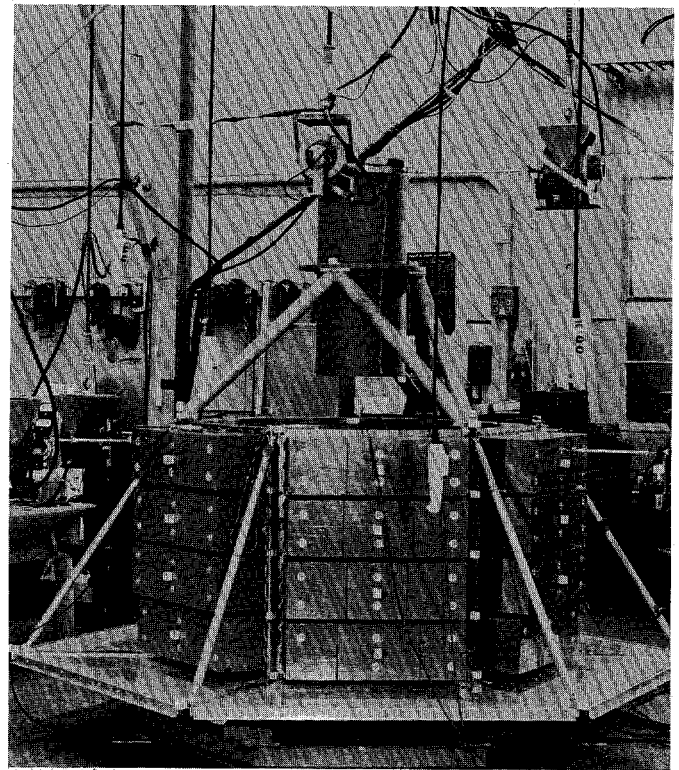


Figure 9. Modal vibration test setup, Mariner B analytical model without fold-out solar panels

The model hex bay was programmed to start at steady-state in normal cruise position. A constant velocity turn of 180 deg/hr of ½-hr duration, caused the face of the bay to become normal to the sunlight. In this position, the bay was allowed to come to steady-state conditions, then programmed back to the cruise position at the same rate. Module temperatures were 85°F at the start of the turn, 140°F after 2 hr, and 190°F after 7 hr normal to the sunlight. The latter temperature is steady-state.

In light of this information, a study was made of the control requirements for the gyro package, the package which has the tightest temperature tolerances aboard the vehicle. Each gyro has a small servo actuated heater capable of doubling the power dissipation if necessary. It has been determined that these heaters, due to the high temperature rise during a worst case maneuver (discussed above), are inadequate for maintaining the gyros within their tolerance.

Since the attitude of the vehicle during a maneuver cannot be predicted, additional restrictions appear necessary at this time. These restrictions may include any or all of the following:

- (1) Additional active controls in series with the heaters.
- (2) Restricting the roll orientation during a maneuver.
- (3) Limitation on the maximum time of a maneuver.

Investigations are currently underway to determine the most adequate restriction or combination of restrictions to use.

b. Planetary horizon platform (PHP). The PHP temperatures during encounter are sensitive to the trajectory. Since the trajectories are still unknown, the preliminary analysis has been based on extreme possible cases of Sun input and configuration factors. These extremes may or may not occur, depending on the trajectories flown.

For the Venus package, first preliminary results show that: the microwave radiometer can be adequately controlled passively, the ultraviolet spectrometer will require active controls, and some changes, as yet undetermined, are needed to adequately control the horizon scanner. Data is now being obtained for more refined calculations.

If the PHP is stowed during cruise, it will require power to keep it warm. The possible power sources are electric heaters, radiation from the bus, sunlight through holes in the fixed solar panel, and radiation from uninsulated fixed solar panel. The amount of power needed can be reduced by a movable insulation blanket around the PHP. A combination of these factors is possible. Another possibility for cruise is to move the PHP out into the Sun, though the details have not been determined.

Similar first results for the Mars package are not yet available. However, it appears that if the Sun is within 90 deg of the planet, the instruments will exceed their maximum allowable temperatures.

c. Capsule. Two different approaches to capsule temperature control have been considered. First, it may be possible to insulate all of the capsule outside surfaces with a thick layer of superinsulation, such as to cut heat losses drastically. Then, by balancing the equilibrium temperature of the capsule around the mid-point of the coasting trajectory, internal temperature variations are expected not to exceed the established limits (-20 to 100°F; unactivated battery).

A second solution proposes covering part of the capsule surface with solar cells and feeding the output to a thermostatically controlled heater inside the payload

package. The insulation requirements would not be as exacting as in the passive control system and would not put as severe structural constraints on capsule and antenna.

The suitability of a *Discoverer* capsule for an early *Mariner* B mission is being investigated by General Electric under a JPL work statement. A slightly larger capsule ($D = 3.2$ ft) with two optional power levels for the transmitter and 40-w instrument load has been studied as an alternate for the *Discoverer*. Power supplied by solar cells covering a part of the capsule surface would both recharge the storage batteries for the transmitter and provide energy for the internal payload heater. For this case, a heat barrier type insulation is not feasible since it would not permit rejection of heat accumulated during the transmission or battery charging periods. However, it may be possible to achieve a satisfactory thermal balance by use of moderate amounts of insulation on the capsule payload and by supplementary electric heaters with thermostatic control.

Although the permissible duration of transmitter operation increases with capsule distance from the Sun, the energy available for any given amount of solar cells decreases, and thus limits the communication.

Capsule-bus interface conditions will markedly depend on the capsule temperature control system selected for the *Mariner* capsule. The interface design will probably be an adiabatic barrier between the capsule and the bus, which would minimize thermal effects on the bus-capsule separation for both bodies. The actual temperature control design required for the separated capsule will determine if additional capsule heaters and insulation surrounding the capsule are required prior to separation.

5. Instrument Integration

Sufficient information has become available on the instrument view requirements, geometry, temperature tolerance, and approximate vibration tolerance to assign preliminary locations on the spacecraft for the 1964 experiments. All alternate locations (that are now apparent) which satisfy each experiment have also been selected, since the optimum locations are related to joint occupancy with other systems, cg control, moment and product of inertia control, cabling, thermal control, and other detail design parameters.

The spacecraft locations which have been used, listed in the order of preference, are:

- (1) Packaging area on the hex bays.

- (2) Structural ring above ($-z$) the hex bays.
- (3) Fixed antenna boom.
- (4) Solar panel tips.

The two experiments, whose look directions require a specific orientation to the ecliptic plane, pose a difficult problem since the spacecraft roll angle to the probe-Canopus line has not been determined. From the preliminary layouts that have been made, one of these experiments can only tolerate roll angles ($y-y$ to Canopus) of 0 to 15, 40 to 60, 100 to 120, and 165 to 180 deg. The other can tolerate 30 to 60 and 120 to 150 deg.

The planet-oriented experiments continue to present problems related to a spacecraft roll orientation to Canopus compatible with the view requirements of the antenna. From the trajectory and planet pass computer runs which have been made, it is apparent that a single spacecraft roll attitude to the probe-Canopus line cannot be used for all flights nor for the whole firing period for each flight. The spacecraft vs antenna and PHP restraint studies are continuing as further trajectory and planet pass information becomes available.

6. Materials

a. *Four-ft high gain antenna.* Some alternate design and fabrication techniques are being investigated in an attempt to manufacture an improved lighter antenna

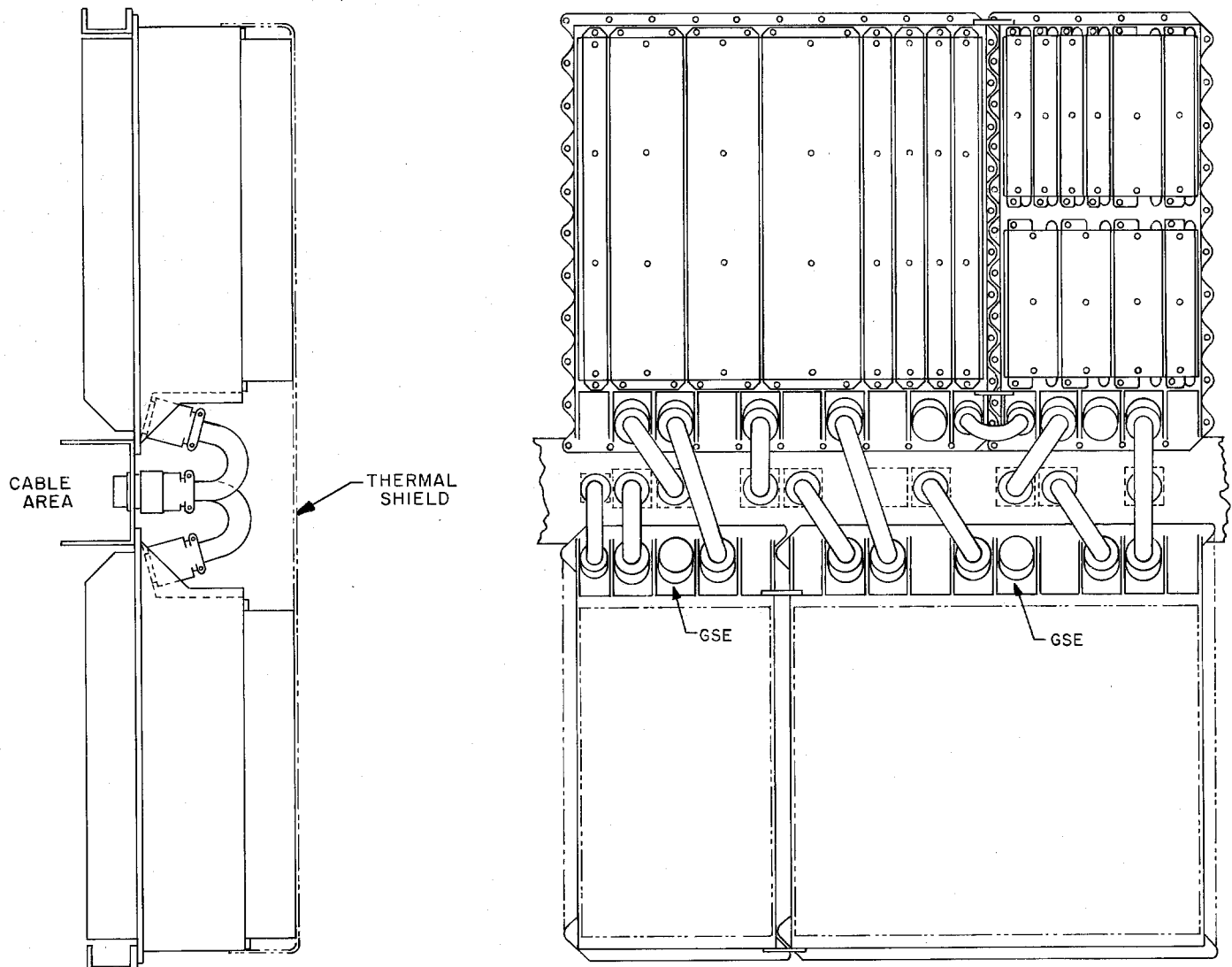


Figure 10. Typical bay showing cable trough

compared to the *Ranger-Mariner* R dish. Efforts are being made to maintain an integral reflector and structure.

One proposal incorporates an explosive formed solid sheet reflector with adhesive bonded or spotwelded stiffeners and attachment reinforcements. A lightened structure results by chemically milling recessed areas on the back side in a pattern that still produces required radial and circumferential rigidity. If thermal considerations require, the dish could be perforated and probably satisfy the structural and electrical requirements.

Sandwich-type construction might involve a furnace-brazed, preformed metallic honeycomb core and facings (solid or perforated). Vendor capability exists in this area with long lead time. Epoxy bonded honeycomb has less joint heat transfer and is not as desirable.

b. Solar panels. A program is being initiated to evaluate ultraviolet-vacuum degradation of cover glasses and adhesives, utilizing equipment constructed at Hughes for paint evaluation. The equipment is readily available for this program.

7. Packaging

In order to accomplish system checkout without removing an assembly from the structure, a cable junction box will be placed between the assembly and the system cabling. Incorporation of the junction box capability resulted in a considerable change in general packaging design.

By combining two of the previous $\frac{1}{4}$ bay assembly increments, the larger cable space obtained would accommodate a 22 shell size connector of the Pygmy-type. This provided an assembly cross-section of the same dimensions as that of the *Ranger* assemblies, with a shelf for the assembly connectors required for the interconnections to the system and ground support equipment. By symmetrically arranging each bay with assemblies above and below a 3.5-in. wide centrally located cable trough, with the assembly connector shelf adjacent to the trough, the cable breakout capability was realized. Figure 10 shows a typical bay with this arrangement. Figure 11 shows a mockup of a portion of the cable section using Pygmy connectors. The case harness used in each assembly will have hard mounted system and GSE connectors. Connection to the system could then be made from the ring harness or from hard mounted connectors installed in the cable trough. The system harness has a capability of 13 Size 22 shell and 2 Size 14 shell Pygmy hard mounted connectors per bay.

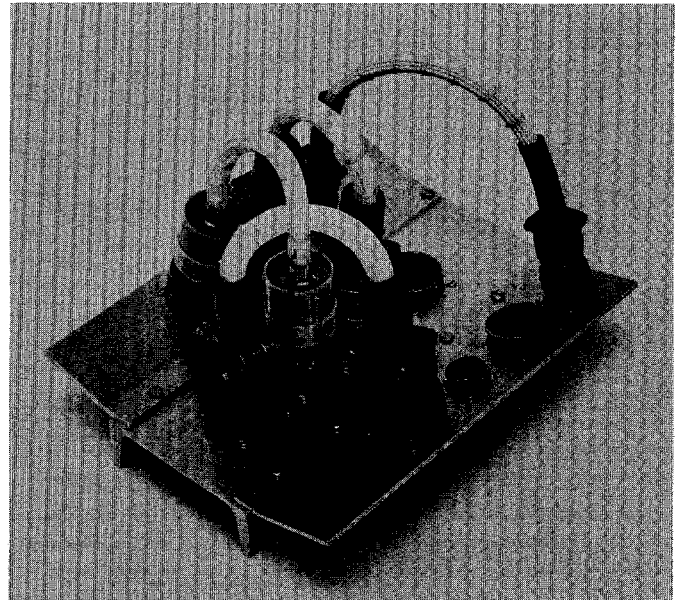


Figure 11. Cable section mockup

All assembly chassis now have a uniform cross-section whether they are located above or below the ring harness. This cross-section will accommodate two rows of the nominal 6- by 6-in. modules (or subassemblies) or one row of 6- by 14-in. wide subassemblies. Capability exists, by utilizing jumper cables, for accomplishing direct interconnections between adjacent assemblies without connecting into the system ring harness. The ability exists to make an assembly any width, in increments of 1.75 in., so long as it is compatible with its adjacent assembly. However, the first modular breakdown will attempt to utilize $\frac{1}{8}$ bay width (8.75 in.) increments.

Figure 12 shows the latest version of the system packaging layout.

8. Entry Capsule

A 3-month study contract has been let to the General Electric Company, Missile and Space Vehicle Division, to determine the suitability of the *Discoverer* and *NERV* entry vehicles for entry through the atmosphere of Mars. Reasons for considering these existing designs for a possible entry capsule mission in 1964 are to: (1) reduce cost, time, and manpower requirements, and (2) utilize a flight tested design system.

The study, in brief, will result in the determination of permissible useful weights (payload) for meeting specified performance requirements and other constraints.

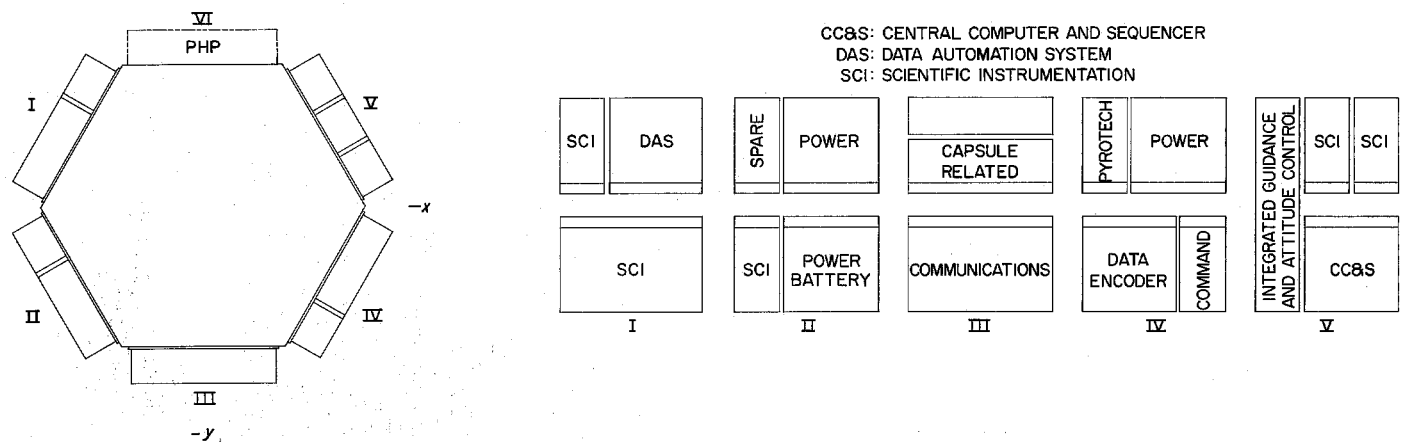


Figure 12. Mariner B system packaging layout

Modifications to the existing design, as may be required, will be described. Specific areas to be examined are the: (1) heat shield, (2) structure, (3) temperature control of the instrument package, and (4) parachute system.

The first preliminary estimates of performance for the two entry vehicles can be summarized in Table 2 for a vertical entry angle, the most adverse atmospheric model, and an initial parachute deployment at Mach 2.2.

Table 2. First preliminary estimates of performance

Vehicle	$m/C_D A_r$ slugs/ft ²	Altitude of completion of parachute deployment, ft	Total weight, lb	Payload weight, lb
Discoverer	1.15	25,000	154	25
	1.48	10,000	198	67
NERV	1.15	25,000	51	8
	1.48	10,000	66	23

IV. Voyager Project

Voyager missions are envisioned to evolve to sophisticated interplanetary spacecraft to be used as planetary orbiters and landers and out-of-the-plane-of-the-ecliptic probes beginning in 1966 or 1967 using the *Saturn* vehicle as the booster.

Planetary orbiters are a necessary part of the evolution of planetary-interplanetary exploration and are a logical step in the development of spacecraft technology. Orbiters will be required to permit sufficient observation of the planet to know how to proceed with the landing missions and may also participate in the actual landing experiments. The emphasis on planetary orbiters, therefore, will be to contribute to the successful accomplishment of the landing objective.

Specific efforts directed to accomplishment of the *Voyager* Project were initiated at the start of FY 62. A brief synopsis of these efforts is as follows:

The Systems Division is participating with the technical divisions and the Marshall Space Flight Center in generation of mission objectives and preparation of preliminary systems design specifications.

The Space Science Division is conducting *Voyager* scientific feasibility studies, establishing the feasibility of meteorological, biological, and certain adaptations of geological experiments for the *Voyager*, as well as refining our knowledge of the Mars and Venus environments as they affect experiments and spacecraft for these flights.

The Guidance and Control Division is continuing its studies of guidance and control requirements, methods, and techniques for advanced planetary missions, both those currently planned and those that become necessary or desirable as a result of discoveries on early planetary exploration missions. The methods to be used include study of possible mission requirements to deduce the general requirements for guidance and control subsystem performance, evolution of subsystem configurations to accomplish the required tasks, and parametric and error analyses of typical missions and operating situations to determine the ranges of parameters and tolerances required.

The Engineering Mechanics Division is performing *Voyager* spacecraft studies, including: analytical investigations of configuration, thermal control, entry and landing dynamics, postlanding and roving techniques, booster compatibility, multiple mission or universal design concepts and compromises, and structural design problems.

In addition to the technical preparation underway within the technical divisions, effort is being expended in the Planetary Program Office in the preparation of a set of mission objectives for the *Voyager* project. These objectives are intended to serve as a guide for a *Voyager* Study Committee to be formed in the near future for the purpose of generating design criteria for use in the preliminary design phase. The primary current problem is the determination of which launch vehicle will be assigned to the *Voyager*. Detailed project planning cannot, of course, proceed until this decision is made.

PART FOUR

ADVANCED DEVELOPMENT

I. Lunar

The Lunar Advanced Development Program consists of those projects and efforts which will extend the state-of-the-art and will provide knowledge applicable to a specific problem or technical field of interest, thus leading to the development of hardware necessary to the planned Lunar Exploration Program.

The Advanced Development Program also establishes a mechanism to ensure exploitation of advances in the state-of-the-art.

A. Temperature Control

Work is continuing on the development of stable white coatings at Armour Research Foundation under JPL contract. Screening tests of candidate pigments show severe ultraviolet degradation of many stable materials such as Al_2O_3 , ZrO_2 , SiO_2 , MgO , BN , and CaSiO_3 . Of the pigment materials tested to date, only ZnO and ZnS show sufficient stability to be potentially satisfactory pigments.

II. Planetary-Interplanetary

A. Synopsis

The Planetary-Interplanetary Advanced Development effort during FY 62 includes the following long-term studies, analyses, and developments leading to feasibility demonstrations of advances in state-of-the-art techniques intended for application to future planetary-interplanetary projects beyond the first flight missions now scheduled.

The JPL Systems Division is conducting systems analysis studies related to trajectory and guidance problems associated with the return of a vehicle from a planet to

the Earth; studies of trajectory and guidance analysis problems for missions to planets other than Venus and Mars; studies in planetary entry and descent from planetary orbits; and studies of extended trajectories to other planets. In addition, studies are being conducted to determine the values and effects of errors in the fundamental solar system constants.

The Space Sciences Division is developing an exobiology microscope, a soil sample gathering microscope vidicon system capable of performing visible analysis on soil samples which have been graded according to particulate size and density, and a companion microscope

spectrophotometer to provide a spectral scan in the range 2000 to 5000 Å of a sample located on the stage of a microscope. Additional efforts are directed to developing sample handling systems for the above equipment. Also underway are developments of microwave receivers to provide research tools for the scientific study of lunar and planetary surfaces, atmospheres, and ionospheres, development of instrumentation utilizing solid state radiation techniques for making measurements of fluxes and energy spectra of charged particles in space, study on both a theoretical and an experimental basis of the merits of the interferometer type of infrared spectrometer, and a study of wet chemical analysis techniques applicable to biochemical analysis and life detection.

The Telecommunications Division is pursuing the development of a planet-based capsule communication system; the study, design, fabrication, and testing of erectable large spacecraft antennas; the investigation of the application of all magnetic logic circuitry to spacecraft time-multiplexing and programming subsystems; the development of a high-capacity data storage system; the evaluation of various methods of range acquisition; and the analysis and development of various circuit techniques applicable to long range communications.

The Guidance and Control Division is conducting advanced development efforts in the area of logical designs for guidance computers; developing experimental models of 9-ft concentrators and a 500-w thermionic converter to provide techniques which may prove feasible as spacecraft power sources; and surveying and developing advanced attitude control system mechanization methods and the various design techniques unique in their implementation.

The Engineering Mechanics Division is performing advanced development of the use of crushable structures for entry capsules; the thermal stress plastic analysis of radiator structures; the development of inflatable instrumentation booms for spacecraft application; the study of atmosphere entry and temperature control aspects of various spacecraft configurations; and the development of techniques applicable to the environmental survival of equipment.

The Engineering Facilities Division is developing instrumentation components, equipment, and systems to support the advanced development efforts of the other divisions.

In the field of chemical propulsion, the Propulsion Division is conducting an industry-wide survey of high energy solid propellant development activities in order

to select a few propellants for more thorough investigation for use in spacecraft propulsion. Efforts are also being expended in the evaluation of throat and nozzle materials.

In the field of electric propulsion, the Propulsion Division is keeping abreast of preliminary descriptions of the SNAP-VIII reactor system, its power conversion system, and its thrust units, and is evaluating the constraints they place on the spacecraft and/or the spacecraft should place on these units. Supporting the electric propulsion are the following efforts being undertaken by other divisions: trajectory and guidance analysis; systems studies and preliminary analysis of spacecraft designs; communication analysis; analytical design of guidance and control subsystems; and studies and analyses in the areas of structural dynamics, structural configuration, shielding, and temperature control for electric propulsion spacecraft.

B. Guidance and Control

1. Study of General Controller Problem

A necessary requirement for current spacecraft is the implementation of controller and sequencer functions (on board the spacecraft) to initiate events and sequences of events at controlled times. In order to facilitate the implementation of future spacecraft controllers, a study of the general controller problem has been initiated.

The objectives of such a study are two-fold: First, it is hoped that a sufficiently general statement of controller functional requirements can be obtained from present mission requirements to permit the synthesis of a basic organization. The controller requirements of other missions then might be satisfied by simple changes in the basic organization to increase the functional capacity of the controller. Emphasis on the organization of a basic controller module which can be expanded by adding modules should both facilitate the design of future controllers and enhance the reliability as a result of the additional experience with a proven unit. Second, it is hoped that such a study will facilitate the generation of controller and sequencer programs for an on-board digital computer.

In order to expeditiously exhaust the possible system or computer program organizations that satisfy the controller requirements, it is necessary to logically classify

the possible approaches. It seems, from the list of functional requirements, that the method of selecting a next-time event from memory offers the best key to an initial classification. If subsequent event times are stored in subsequent memory addresses, the event times are said to be ordered. The first consideration in this initial classification has to be whether the event times in memory remain ordered to a large degree or remain ordered to only a small degree. This unpredictable ordering of event times is a result of permitting command channel modification of event times. This consideration leads to the following possibilities:

- (1) Event times remain completely ordered.
- (2) Event times remain ordered to a large degree.
- (3) Event times remain ordered to only a small degree or not at all.

The first possibility can be effected rather simply by loading the memory addresses in sequence and taking the next address as the next time event. Such a solution is not interesting, since the occurrence of completely predictable ordering is not considered likely. The division between a large degree of ordering and a small degree of ordering is, however, more interesting, especially since it is difficult to decide which best described the general controller problem.

Taking the completely unordered case first, the problem becomes one of selecting the next event from memory and executing the associated command at its respective time. Possible ways of implementing this operation can be classified as follows:

- (1) Associate clock with time part of command; execute on successful association.
- (2) Scan time part of commands and compare to clock; execute on successful comparison.
- (3) Scan time part of command and increment time part; execute on zero time part.
- (4) Reorder commands on time modification; execute in order at respective times.

Considering now the partially ordered case, it is obvious that the above techniques apply; however, an interesting alternative is to treat the modifiable commands as in the unordered case and the unmodifiable commands in a separate ordered part of the memory. Execution of commands is then effected by associating or scanning the modifiable portion of memory and the next ordered command each clock time, executing the commands that are

current. Such a scheme will take advantage of partial ordering to reduce data transfers at the possible expense of the total hardware count.

In the case of partially ordered event times, a significant advantage comes not so much from a particular classification of memory search but rather from the resulting division of the memory into a modifiable and a nonmodifiable part. This permits expansion of the non-modifiable portion of memory with no change in organization. Expansion of the modifiable portion is not as simple. However, this is not serious because the useful capacity of a modifiable memory is probably limited by telemetry capacities. Programs for the generation of controller functions can also take advantage of the partial ordering concept by processing the memory as separate lists.

2. Solar Energy Thermionic Conversion

Most of the effort being sponsored by the Laboratory in the general area of spacecraft secondary power advanced development is related to thermionics. Three contracts are currently being worked on.

The development of a solar energy thermionic (SET) conversion system suitable for use as the source of electrical power on a Mars spacecraft has entered its second phase. The results of Phase I have been described in SPS 37-13. The design objectives of the SET system are 1-year service life, 15% system efficiency at Mars, 135 w of electrical power throughout transit from Earth to Mars, and a system weight of 25 lb. The system will incorporate a 5-ft diameter solar concentrator, a solar flux control mechanism, and five cesium-vapor-filled thermionic energy converters mounted on a cubical generator which is supported at the focal point of the concentrator. Electro-Optical Systems, Pasadena, Calif. (EOS), which is the prime contractor, will develop the solar concentrator, generator supports, and solar flux control mechanism. EOS will participate in the solar cavity development and will perform electrical tests on thermionic converters and generators using electrical and solar heat sources. The development of the SET generator has been subcontracted by EOS to Thermo Electron Engineering Corporation (TEECO), Waltham, Mass.

The Phase II effort provides for the performance of four specific tasks. The objectives of Task 1 are the experimental verification that:

- (1) The series SET converter can start, cycle, and operate in any position.

- (2) The metal-to-ceramic seal can withstand the anticipated generator environment (namely, the 600°C operating temperature).
- (3) Transverse forces will not decrease converter life to less than 150 hr when the converter operates in a horizontal position.
- (4) Repeatable converter power output of at least 12 w at 1 v can be achieved.

In addition, TEECO will deliver two Series VI converters to EOS and one to JPL for performance evaluation. EOS and JPL will independently obtain current-voltage characteristics at various emitter, cesium reservoir, and radiator temperatures so that optimum values of these parameters can be determined for the Series VI converters.

Upon satisfactory completion of the first task, work on Task 2 will be initiated. The objectives are:

- (1) Fabrication of Thermionic Generator JG-1.
- (2) Preliminary laboratory evaluation of generator performance using electron bombardment heating.

The objectives of Task 3 are:

- (1) Measurement of the temperature distribution in a heat transfer model of the SET generator.
- (2) Identification and solution of problems associated with the solar test of thermionic generators.
- (3) Measurement of the individual performance of the five converters assembled in Generator JG-1 using a SET mirror as the energy source.
- (4) Measurement of the performance of Generator JG-1 with the five converters connected in series using a SET mirror as the energy source.
- (5) Evaluation of Generator JG-1 performance using an electron bombardment heater.

Two of the Series V SET converters will be examined in Task 4 in an attempt to identify possible life limiting factors which may be discernible in these units. The Series V converters are of particular interest due to the operating lifetimes achieved with these units; i.e., Converter Va operated 850 hr without failure, Vd failed after 421 hr and 137 thermal cycles.

The Series VI SET converters, which TEECO will fabricate under the EOS subcontract, will have tantalum emitters and emitter supports. It is anticipated that these converters will produce approximately 6 w/cm² at an emitter temperature of 1700°C and 1-v output. Analysis indicates that a power density of 14.4 w/cm² at 1 v is

required to obtain the design objective of 20% efficiency for the SET thermionic generator. TEECO, under a JPL contract, will investigate rhenium and tungsten to determine whether either of these materials will produce more power (w/cm²) than tantalum at 1700°C emitter temperature and 1-v output. Six converters will be fabricated: three with rhenium and three with tungsten emitters. The converters will be identical to the Series VI SET converters except for the emitter. TEECO will deliver one converter of each type to JPL for performance evaluation.

EOS has been awarded a contract for the investigation of materials and techniques applicable to thermionic energy conversion. The investigation will include (for each of six tasks) a survey of the technical literature, both foreign and domestic.

Task 1—Emitter, collector and construction material, and fabrication techniques. EOS will survey the areas of emitter materials, collector materials and construction materials. Features to be investigated include: (1) electrical properties, (2) thermal and physical properties, (3) chemical and metallurgical properties, and (4) mechanical properties.

Task 2—Ceramic-to-metal seals. State-of-the-art processes involving metal-to-ceramic seals pertinent to the thermionic converter technology will be investigated. The features of metal-to-ceramic seals to be studied include: (1) vacuum tightness, (2) reliability, (3) cesium compatibility, (4) high temperature operation, (5) strength, and (6) thermal shock.

Task 3—Heat transfer. Methods used in determining the heat losses incurred in thermionic energy converters of practical design will be evaluated. Accurate techniques for obtaining the heat loss resulting from electron cooling, plasma conduction, radiation, and conduction through mechanical structures will be formulated and verified where possible.

Task 4—Electrode spacing techniques. The high perveance diode and high transconductance triode electron tube art will be surveyed, and those features pertinent to thermionic converter spacing will be extracted. A survey and evaluation shall be performed of the present status of electrode spacing techniques used in the field of thermionic energy conversion, and promising techniques shall be identified.

Task 5—Physics of thermionic energy conversion. The physical phenomena basic to thermionic energy conversion will be analyzed. Thermionic emission and ionization processes are specific examples of such phenomena. With the aid of data in the literature relating to thermionic

converters and gas discharge devices, an attempt will be made to formulate a mathematical model (or models) which applies to the vapor thermionic converters over a specific region of operation.

Task 6—The properties of cesium and its use in thermionic converters. The electrical, optical, chemical, and thermodynamic properties of cesium as exist in the literature will be analyzed. Particular attention will be given to cesium compatibility information generated in studies of thermionic converters, ion engines, and gas discharge devices. The optical and electrical properties of cesium (which are basic to an understanding of step-wise ionization through the process of the capture of resonance radiation), will be very carefully studied.

3. Solar Cell Evaluation

The combined balloon and electronics groups of General Mills, Minneapolis, Minn., have been awarded a contract for the performance of three solar cell evaluation balloon flights. The purpose of these flights will be to observe the performance of solar cells in what are essentially space conditions. All three flights will be conducted at a maximum altitude of 80,000 ft. This altitude will put the cells far enough above the atmosphere to reduce the attenuation of sunlight due to atmospheric scattering to 1% of what it is at sea level. The water and ozone absorption bands will also be eliminated for all practical purposes.

The main function of these flights will be to measure the short circuit current from different cells. The current measurements from the balloon flights will be compared with current measurements on the same cells at Table Mountain, California. By comparing the Table Mountain data with balloon flight data, a space extrapolation factor for the output of cells at Table Mountain to the output in space may be obtained. Such a factor is extremely important in predicting the output of solar cell power supplies based on Table Mountain tests. The data from the balloon flights will serve as a check on the extrapolation factor presently being used. The total inaccuracy of the efficiency degradation factor obtained with balloon data will be $\pm 2\%$ plus the inaccuracy due to solar irradiance measurements. The total inaccuracy in the degradation factor obtained from balloon flight data should then be less than $\pm 4\%$.

The cells to be measured will be mounted on a Sun tracker which, in turn, will be mounted on top of the balloon. The output of the cells will be fed into a voltage controlled oscillator, the output of which will be transmitted on an FM-FM telemetry system. The telemetry

data will be recorded with both an oscillograph and a counter-printer clock. There will be inflight calibration of the telemetry to eliminate errors due to drift in the system. The over-all accuracy of the telemetry should be better than $\pm 2\%$. The first balloon flight is scheduled to take place the week of March 5, 1962.

C. Engineering Mechanics

1. Frequency Response Data for a Shallow, Spherical Shell

A limited amount of analytical work has been reported in the literature on the dynamic behavior of shell structures; however, the work to date is extremely theoretical and restricted to particular problems that lend themselves to the analytical methods available. Therefore, a brief experimental program was undertaken to obtain a modal survey for a shallow spherical shell supported at the apex. This particular system was chosen to simulate a parabolic antenna supported in the same manner. The shell dimensions and mounting technique are shown in Figures 1 and 2.

The shell segment was mounted to a shake table through the rigid insert and subjected to a sine wave frequency sweep from 10 to 200 cps. The acceleration input during the sweep was varied such that the modes were clearly determined:

Modal patterns	Natural frequency, cps
Four-node mode pattern; large deflections for low energy input	19 (axial)
Rigid body rotation of shell about insert; large deflections for low energy input	35 (transverse)
Six-node mode pattern; large deflections for low energy input	43 (axial and transverse)
Eight-node mode pattern; small deflections for large energy input	79 (axial)
Ten-node mode pattern; small deflections for large energy input	119 (axial)
Eight-node mode pattern; large deflections for low energy input	150 (axial)
Symmetrical extensional mode	195 (axial)

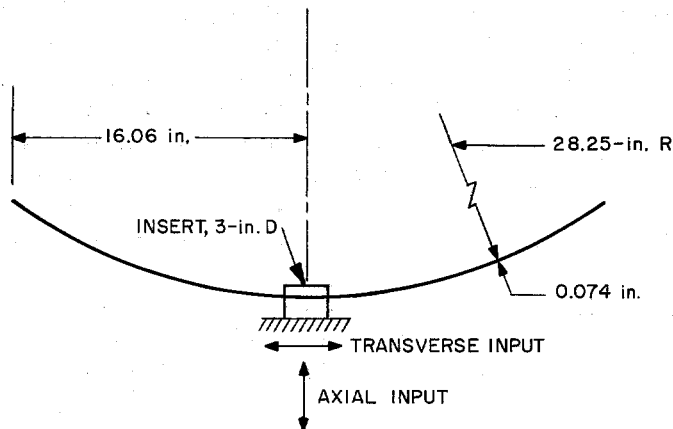


Figure 1. Shell dimensions

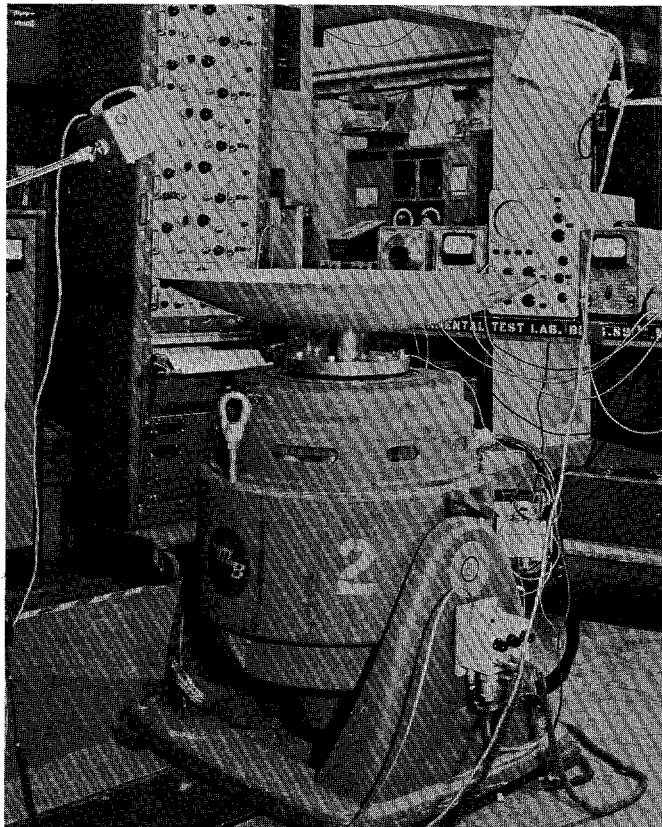


Figure 2. Shell mounting technique

Three types of modal patterns were observed: (1) the n -node, (2) the mode due to local flexure about the insert, and (3) the axial symmetric extensional mode. These modes are illustrated in Figure 3.

The resulting modal patterns thus provide means by which an orthotropic shell may be designed such that

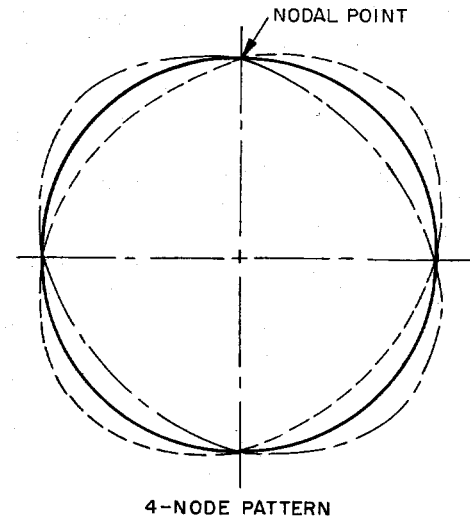


Figure 3. Modal patterns

the fundamental frequencies are increased by effectively stiffening the shell structure.

2. Vidicon Tube Shock Tests

As part of the vidicon testing previously reported in SPS 37-11, five ruggedized 1-in. diameter electromagnetic vidicon camera tubes (GEC Type 7226A) have been shock tested. The tubes were rejects with defects which were not believed to affect their shockworthiness; they were supplied by General Electrodynamics Corporation on a no-charge loan basis.

The vidicon tubes were mounted in a manner similar to that previously used; that is, each was potted in a close fitting hole in a two part aluminum shock test fixture. The purpose of the potting materials was to conform to the irregularities of the tube envelope, giving the envelope as nearly uniform support as was possible so that the tube

and not the mounting method would be evaluated. Although the Dow Corning RTV 881 silastic used is resilient, it was employed in a thin (0.025 in.) section so that deflections of the tube envelope relative to the shock test fixture are small, the only significant departure from true hard mounting being some damping of high frequencies.

Before each test (or, at lower levels, each pair of tests at one level), the tube was placed in a working TV camera circuit to check its proper functioning. After the test (or pair of tests), the tube was again placed in the TV circuit. Any noticeable picture degradation on the monitor constituted a failure. Actually during this test series, no minor degradations were encountered; all failures noted were complete losses of picture.

The tubes were tested at successively increasing accelerations from 325 to 4000 g at impact velocities of 38 to 110 ft/sec. Axial shocks at 900 g caused a slight shift of the gun structure within the envelope but caused no picture degradation. All axial shocks above this level caused failure. At levels to and including 1700 g, there was no indication of damage due to lateral shocks. One tube which had previously been subjected to a 1700-g lateral shock failed at 2500 g in the lateral direction due to a cracked envelope. To preclude the possibility of cumulative damage, two previously untested tubes were subjected to one lateral shock each. One tested at 2500 g survived with a slightly bent gun structure. One tested at 4000 g at 110 ft/sec failed, apparently due to a broken heater lead.

In conclusion, the 7226A vidicon tube apparently will withstand lateral shocks of about 2000 g. This tube is constructed in a manner that is intended to mechanically isolate the gun from the envelope to provide vibration immunity during operation but does not constrain the gun axially; thus, in this direction, the failure level is of the order of 1000 g. The manufacturer has indicated ways in which this could probably be remedied.

D. Propulsion

1. Amine Nitrate Monopropellants

Liquid monopropellants currently serve as energy sources for spacecraft in-transit trajectory correction devices. A 50-lb-thrust rocket motor utilizing monopropellant hydrazine was recently employed successfully on the *Ranger III* spacecraft for such an in-transit maneuver.

A similar hydrazine propulsion device is planned for other *Ranger* missions as well as for the *Mariner R* and *Mariner B* planetary probes. For future missions, however, system studies indicate the desirability of more energetic liquid propellants since propulsion system weights become appreciable as spacecraft weights increase. A candidate propellant for such applications is the high energy liquid monopropellant designated Cavea-B. Experimental investigation of this propellant at the Laboratory, conducted for about 2 years, has been chronologically reported in the classified volumes of *Research Summaries* 36-2 through 36-7, and *Space Programs Summaries* 37-9 through 37-13.

Recent experimental tests were made in order to investigate the effects of combustion chamber contraction-area-ratio, ϵ_c , on engine performance. All testing was conducted at the 100-lb-thrust level utilizing propellant at the 1.50 acid-to-salt ratio. The rocket motor configuration consisted of 2.5-in.-diameter silicon-carbide-lined combustion chambers and the impinging stream injector previously described in *SPS 37-10*. This injector provides propellant atomization through a circular ring of 12 like-on-like doublet orifice pairs. Ignition was accomplished by the hypergolic reaction of unsymmetrical dimethylhydrazine with Cavea-B.

Previous investigations have shown that a characteristic combustion chamber length L^* in excess of 100 in. is required in order to obtain reasonably high percentages of combustion efficiency (96 to 98%). This L^* requirement leads to excessively long combustion chambers if a low ϵ_c is used. Testing was conducted to determine the performance at chamber pressures of 150 and 300 psia and at two ϵ_c for each pressure: 6.9 and 10.8 for the 150-psia case, and 14.2 and 21.4 for the 300-psia case. Preliminary test results are presented in Figure 4. Additional results will be presented in subsequent reports since one additional ϵ_c will be investigated during the next reporting period. The additional data is expected to produce clarifying results. Most of the data presently available indicates that larger ϵ_c values result in lower performance at any given L^* . Thus, a length parameter as such must play an important part in the combustion process in addition to the residence time necessary for the reaction kinetics to proceed to completion.

2. Advanced Liquid Propulsion System

The Advanced Liquid Propulsion Systems (ALPS) Program has been established for the purpose of formulating and demonstrating solutions to problems in spacecraft propulsion. The program has been organized to

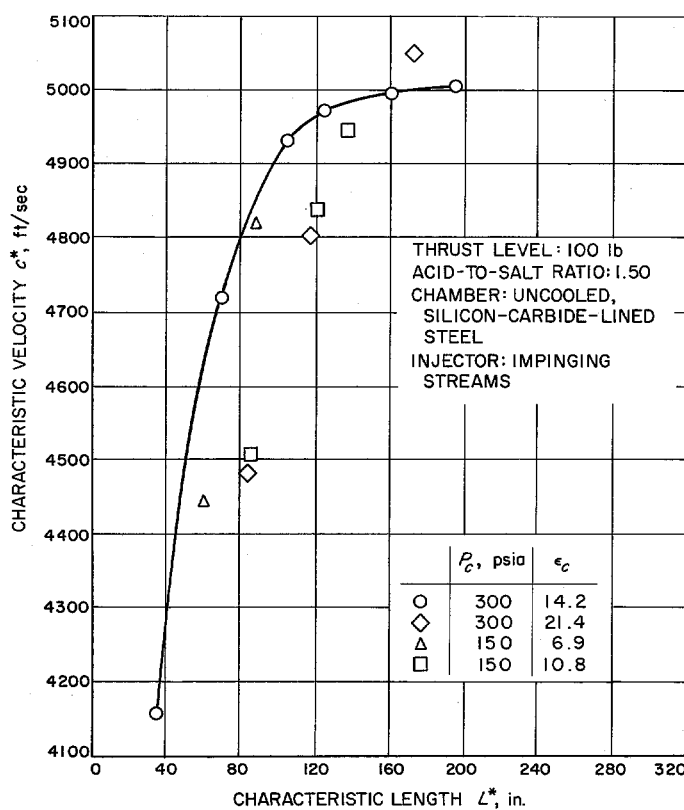


Figure 4. Experimental performance of Cavea-B showing contraction-area-ratio (ϵ_c) effects

cover these problems in a coordinated manner so that the solutions are mutually compatible and suitable for practical application in a complete system. SPS 37-8 through 37-13 contain reports on the progress of the program.

a. Bladder development. During free-fall (so-called zero-g conditions), the forces tending to position the liquids in propellant tanks are diminished to the point where random circulation of the liquids through the gas in the ullage takes place. To operate a liquid propellant rocket system under these conditions, it becomes necessary to separate the propellants from the pressurizing gas so that a continuous liquid stream is fed to the rocket engine. This can be accomplished in several ways, the most common of which is by means of an expulsion bag or bladder.

To be effective the expulsion device must be both compatible with, and impermeable to, the propellants. In addition, the expulsion bladder material for the ALPS system must be flexible enough to achieve an expulsion efficiency of 99% or higher and durable enough to withstand several complete cycles of operation. It must also

be capable of fabrication into spherical and cylindrical shapes.

A search for suitable materials is currently being conducted. This search has been directed toward finding materials that are compatible with, and impervious to, nitrogen tetroxide and hydrazine, with only secondary consideration for problems of fabrication because the former factors are judged to be the most critical of the established criteria. Since permeability testing is also an effective screening method for compatibility and since N_2O_4 is the more difficult of the two propellants to contain, material testing to date has been limited to permeability tests with N_2O_4 (Ref 1). So far, approximately 100 samples of various materials and combinations of materials have been tested. Included were: FEP Teflon, TFE Teflon, FEP-TFE laminates, metal-plated Teflon, FEP Teflon-aluminum foil laminates both heat and adhesive bonded, TFE dispersion cast on aluminum foil, Mylar, Aclar, Kynar, impregnated Teflon, and laminates of FEP Teflon and butyl rubber. None of these materials has proven entirely satisfactory.

It can be stated, however, that laminates of FEP Teflon film in thicknesses down to 0.001 in. and aluminum foil down to 0.0002 in., either heat or adhesive bonded, can be considered an effective barrier to N_2O_4 so long as the aluminum foil is free from pinholes. Adhesive bonding is not acceptable because pinholes in the aluminum allow the N_2O_4 to chemically attack the adhesive thus spreading the damage. The heat bonded laminates composed of 0.0005- to 0.002-in. aluminum foil and 0.001- to 0.005-in. FEP Teflon seem both tough and flexible enough to be considered for bladder construction provided that the following problems of fabrication can be solved:

- (1) Inspection of the aluminum foil for pinholes and other defects prior to lamination.
- (2) Formation of the laminate into spherical shapes.
- (3) Joining the formed segments with a gas-tight metal-to-metal seam of adequate flexibility.
- (4) Reducing, by controlled folding, the tendency of the material to rupture as it flexes during expulsion and filling operations.

These problems are currently being studied by Swedlow, Inc. of Los Angeles. Some success is evident although a completed bladder has not been delivered.

The ultrasonic welding technique was investigated as a means of fabricating the seams where the segments of a Teflon-aluminum foil laminate bladder are joined. Tests with pieces of 0.001-in. thick aluminum foil have proven

that a continuous metal-to-metal weld can be made that is impervious to N_2O_4 within the limits of the test method used (Ref 1). The problems inherent with this fabrication process will become more apparent when it is applied to the construction of bladders, because the welding must be done between a weld head or horn, and an anvil located beneath the work. The method of positioning and manipulating this anvil while welding the closing seam of a bladder has yet to be determined.

Another promising material derived from the program is a multiple laminate of TFE-FEP sandwiched with several layers of chemically applied gold plate. This fabrication process is being developed under JPL contract by Joclin Manufacturing Company of Wallingford, Conn. The combination laminate has reduced permeability rates from those of cast TFE film by a factor of 10 and appears to have the possibility of another equal improvement. Although this reduced rate is still far from the arbitrary standard of permeability established for ALPS, it is considered significant. Because this material is built up by a series of spraying and plating operations it can readily be fabricated into seamless bladders, and it appears to be both tough and flexible.

Single layer laminates of plated metal on Teflon, whether the metal plate is homogeneous or composed of various combinations of metals, have proven unsatisfactory. Results have shown that a plate thick enough to be impermeable to N_2O_4 is too stiff for use as a bladder material. Also, the plate tends to crack and separate when flexed. Nickel plate is attacked by N_2O_4 and will be removed from the Teflon in thin layers. This action is probably due to the HNO_3 formed by combining N_2O_4 with moisture in the atmosphere. Vapor deposited gold and aluminum are removed from Teflon by N_2O_4 and appear to offer little resistance to N_2O_4 penetration. Mylar, Kynar, and Aclar are unsatisfactory either from the standpoint of compatibility, permeability, or combinations of both. Some high density FEP Teflon samples look favorable but further testing will be required to establish their suitability.

b. Effect of prepressure level and ullage fraction upon the pumping efficiency of a stored gas system. It has been the practice at this Laboratory to present the results of gas-pressurized propellant tank expulsion tests in terms of the dimensionless pumping parameter γ' (Ref 2). This parameter is defined as

$$\gamma' = \frac{V_R (P_i - P_f)}{V_P P_R}$$

where V_R is the volume of the high pressure gas reservoir, P_i and P_f are the initial and final pressures in the

high pressure reservoir, V_P is the volume of fluid expelled, and P_R is the regulated tank pressure at which the fluid is expelled. Conceptually, this parameter is inversely related to the pumping work resulting from a given pressure loss in the reservoir. The use of the γ' parameter involves the tacit assumption that the propellant tank is completely full at the beginning of a test. In actuality, there is usually a small ullage volume present in the tank. The gas in this space is often at a pressure different from the regulated value, and, therefore, some of the reservoir gas must be used to bring the tank up to the regulated pressure. In a series of tests with different values of ullage volume and/or prepressure, this effect will vary accordingly and a comparison on the basis of γ' is not possible. The purpose of the following analysis is to determine the influence of the prepressure level and ullage volume upon pumping efficiency and to present a suitable normalizing technique so that some comparison between tests is possible.

The system studied is presented schematically in Figure 5. In the analysis the following simplifications were assumed:

- (1) There is no tank flexure with pressure.
- (2) The liquid is incompressible.
- (3) Vapor pressure of the liquid is entirely negligible.
- (4) Reservoir gas is the same as that in the tank ullage space.
- (5) The gas is ideal ($Pv = RT$) and has constant specific heats.
- (6) Expulsion of fluid occurs at a constant pressure equal to the regulated pressure.

Choosing the gas in the reservoir and the ullage space as a thermodynamic system, and permitting heat to flow to the gas from the surroundings during the expulsion process, the following energy balance may be presented:

Heat added minus work done = change in internal energy

The work done may be expressed as

$$\Delta W = P_R (V - V_x) = P_R V_P$$

where V is the sum of the volume of propellant expelled and the ullage volume (V_x).

The initial energy of the system is

$$u_1 = c_v (m_i T_i + m_x T_x)$$

where subscripts x and i refer to the initial propellant tank ullage and initial reservoir conditions, respectively.

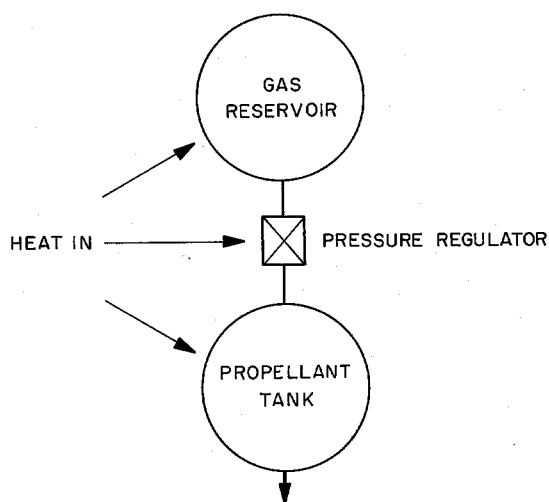


Figure 5. Gas-pressurized propellant tank system

The energy at the end of the process is

$$u_2 = c_v \{m_f T_f + \{m_i - m_f + m_x\} T_o\}$$

where subscripts *f* and *o* refer to the final reservoir and final tank conditions, respectively.

Substituting the preceding into the energy balance yields

$$Q_{in} - P_R (V - V_x) = c_v \{m_f T_f + (m_i - m_f + m_x) T_o - m_i T_i - m_x T_x\}$$

Applying the relations for ideal gases that

$$PV = mRT$$

$$c_p = c_v + R$$

permits the conversion of this equation into the following

$$\frac{c_p}{c_v} \left[P_R V \left(1 - \frac{V_x}{V} \right) - Q_{in} \right] = V_R (P_i - P_f) - V_x (P_R - P_x) - Q_{in}$$

This expression may be cast in terms of the definition of γ' , that is

$$\gamma' \equiv \frac{V_R (P_i - P_f)}{P_R V_P} = \frac{c_p}{c_v} - \frac{Q_{in} \left(\frac{c_p}{c_v} - 1 \right)}{P_R (V - V_x)} + \frac{V_x (P_R - P_x)}{P_R (V - V_x)}$$

The old pumping performance parameter is seen to depend directly on the prepressurization level and the ullage volume, and hence it is inadequate as a measure of performance for systems utilizing large ullage volumes.

A new performance parameter which is more nearly independent of the ullage condition is

$$\beta \equiv \frac{V_R (P_i - P_f)}{P_R V_P} - \frac{\frac{V_x}{V} \left(1 - \frac{P_x}{P_R} \right)}{1 - \frac{V_x}{V}} = \frac{c_p}{c_v} - \frac{Q_{in} \left(\frac{c_p}{c_v} - 1 \right)}{P_R (V - V_x)}$$

This new parameter permits comparisons and predictions of the performance of a given system with various pre-pressure conditions. The heat absorbed by the pumping gas into two different designs is a function of the heat capacity of the tanks, the blowdown rates, the vehicle acceleration level, line size, etc. Hence, a prediction of the β parameter for a particular system must be derived from experiments on similar systems.

Several limiting forms of the β parameter are of interest.

- (1) For zero ullage volume

$$\beta = \frac{V_R (P_i - P_f)}{P_R V_P} \equiv \gamma'$$

- (2) For zero prepressure, as would occur in a vacuum filled system containing a propellant with low vapor pressure

$$\beta = \frac{V_R (P_i - P_f) - P_R V_x}{P_R V_P}$$

- (3) For full prepressure

$$\beta = \frac{V_R (P_i - P_f)}{P_R V_P} \equiv \gamma'$$

- (4) For adiabatic blowdown

$$\beta = \frac{c_p}{c_v}$$

c. Thrust chamber development. Three firings with 100-lb-thrust free-standing pyrolytic graphite thrust chambers were reported in SPS 37-13. During the present reporting period, one additional firing was made, again using the propellant combination $N_2H_4-N_2O_4$. The firing was scheduled for 30-sec duration but the chamber failed at 21 sec, probably because of oxidation of the inner surface of the chamber. This chamber had been used previously in two firings of 30-sec duration each. Additional firings will be made soon using chambers similar to those described.

[REDACTED] **UNCLASSIFIED**

References

1. Vango, S. P., "Determination of Permeability of Cast Teflon Sheet to Nitrogen Tetroxide and Hydrazine," Technical Memorandum No. 33-55, Jet Propulsion Laboratory, Pasadena (1961).
2. Powell, W. B., "Gas-Feed Pressure Systems and Pressure Regulators," Project Note 1-29, Jet Propulsion Laboratory, February 28, 1945.

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III. Syncom

The mission, design, and development of the JPL *Syncom* apogee motor were described in SPS 37-13, Volume II. This report covers activities in the areas of propellant development, results of tests with heavy-wall cases and flight-weight nozzles, inert rocket motor structural test programming, flight-weight case development work, and design integration with the spacecraft.

1. Propellant Development

The propellant development phase of the JPL *Syncom* motor development program is essentially completed with the exception of possible minor modifications in the propellant as the *Syncom* Program progresses. A propellant has been developed which will meet the requirements of this motor. Several 120-lb batches have been made,

thereby establishing the proper processing and curing methods. The propellant, a polyurethane-ammonium perchlorate-aluminum system, currently designated SYN 3-3, has the following formulation:

Ammonium perchlorate, 64.00%.

Aluminum powder, 16.00%.

Polyurethane binder, 19.71%.

Neozone-D, 0.25%.

FeAA catalyst, 0.04%.

The propellant ballistic properties have been fairly well established, and the burning characteristics are shown in Figures 1 and 2. Figure 1 shows the K_N vs P_c

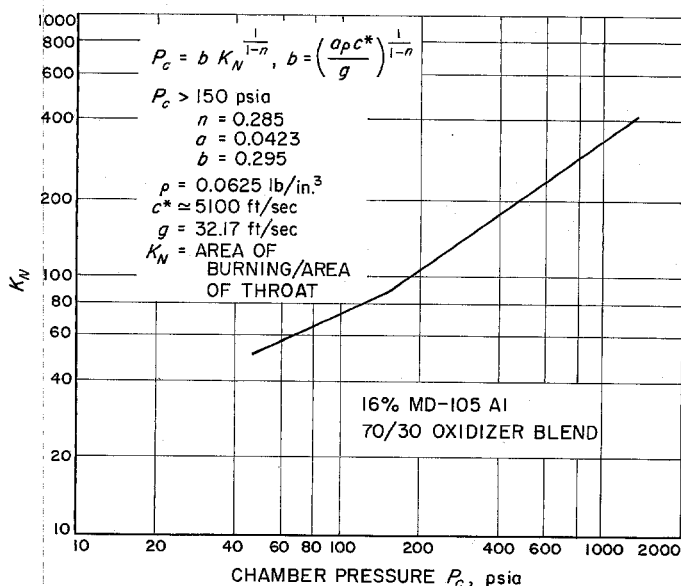


Figure 1. K_N vs P_c of SYN 3-3 propellant at 80°F

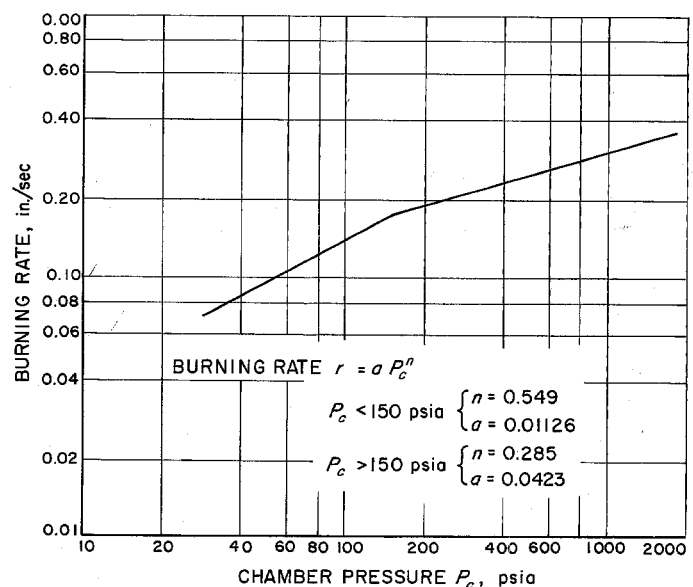


Figure 2. Burning rate of SYN 3-3 propellant at 80°F

plot and Figure 2 shows the 80°F burning rate as a function of chamber pressure. These plots were established by a large number of 5-lb motor firings over the pressure ranges shown. Three constant-pressure motors loaded with 32 lb of propellant have also been successfully fired as well as four heavy-wall *Syncom* motors. The ballistic data from all of these firings agrees with the data in Figures 1 and 2.

The following properties characterize the 80°F physical properties of this propellant: $S_m = 135$ psi, $C_m = 82\%$, and $\rho_p = 0.0625$ lb/in.³ where S_m is the maximum stress, C_m is the elongation at maximum stress, and ρ_p is the density. These tensile properties are based on pulling dumbbell samples at a cross-head speed of 2 in./min which corresponds to a strain rate of 0.74 in./in./min based on the effective gage length.

2. Firing Tests

a. Heavy-wall cases. Because of the long lead-time anticipated for the flight-weight case delivery, inexpensive and readily available cases utilizing 0.25-in. mild steel were fabricated and used for a series of heavy-wall tests. Approximately 55 lb of propellant (compared with 63 lb in the light-weight case) are accommodated in these cases which have a 12.00-in. internal diameter, and 2:1

elliptical head and aft ends on the interior. Insulation materials, ignition systems, grain configuration, and flight-type nozzles can all be basically evaluated in these heavy-wall cases.

Four firings have been conducted to date. A brief compilation of significant items studied in these early development firings appears in Table 1. The firings were all run at 80°F grain temperature, and the pressure-time curves are shown in Figure 3. A typical test setup is shown in Figure 4. The c^{**} (characteristic velocity based on measured chamber pressure and average values of throat area for each run) as shown in Table 1 fall within the expected range of values. One pressure gage read the head-end pressure and two gages read the nozzle-end pressure. Thrust was measured, but the data is not reported because of the flow-separation occurring through the entire run since the nozzle expansion ratio is 20 (designed for later diffuser tests). An exhaust system was utilized to help keep the test pit clear of exhaust which clouds the motion picture photography.

Ignition problems were revealed in this first series of tests. The delay encountered in Runs X-1152, -1166, and -1177 were solved when aluminum-potassium perchlorate (Alclo) pellets were substituted for boron-potassium nitrate pellets in Run X-1185. The large initial pressure

Table 1. Summary of heavy-wall tests

Run	Test date, 1962	Ignition materials (pellets; basket material)	Propellant	Insulation	Nozzle throat material	c^{**} , ft/sec
X-1152	Jan 29	Boron KNO ₃ ; Fortisan	SYN 3-3	Siltemp 84	Mascerated carbon cloth	5100
X-1166	Feb 5	Boron KNO ₃ ; Fortisan	SYN 3-3	Butyl rubber stoner 8111	Mascerated carbon cloth	5170
X-1177	Feb 13	Boron KNO ₃ ; Fortisan	SYN 3-3	Butyl rubber stoner 8111	High-density graphite	5030
X-1185	Feb 23	Aluminum KClO ₄ ; Siltemp 84	SYN 3-5	Siltemp 84	High-density graphite	5010

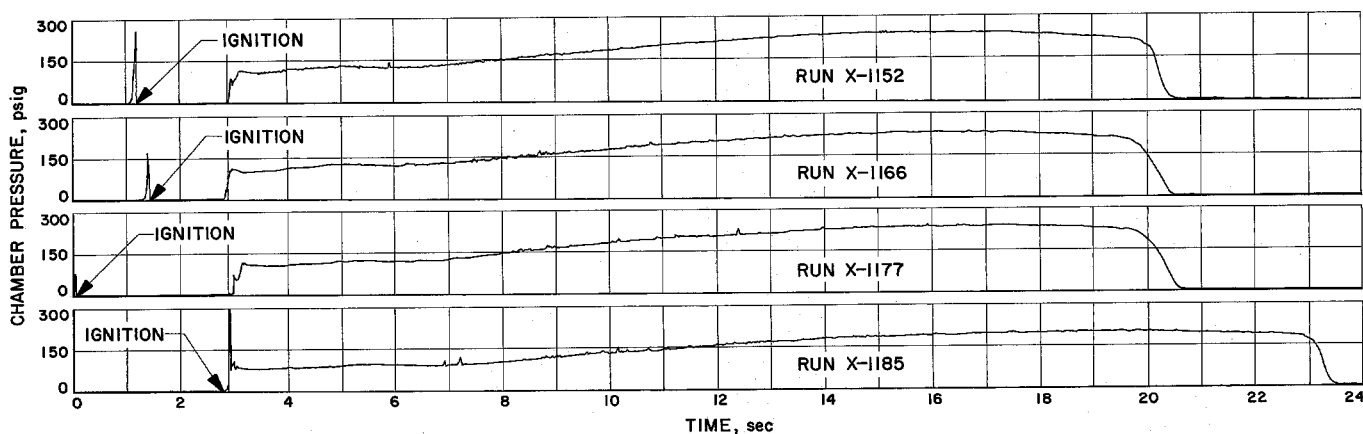


Figure 3. Pressure-time curves for firings at 80°F grain temperature

peak for Run X-1185 is believed caused by blockage of the throat by the igniter basket.

The small pips in the pressure-time curves are believed to be due to the random sloughing off of aluminum, aluminum oxide, and insulation material from the exterior portion of the inserted nozzle facing the propellant grain. There appear to be no detrimental effects from this phenomenon. Later accelerometer measurements now planned may determine the acceptability of this effect.

The motor in Run X-1185 was loaded with the SYN 3-5 propellant, identical to the standard SYN 3-3 except that the oxidizer grind-size ratio (coarse/fine) was changed to 87.5/12.5 from 70/30 for the standard formulation. The attempt was to achieve a higher initial chamber pressure while maintaining the same maximum pressure. Although the burning rate was generally lower throughout the run, the desired pressure ratio was not attained; it has been decided not to change propellant oxidizer blend.

b. Motor design changes. As a result of these first heavy-walled development tests, design improvements are indicated in several areas. Both nozzle designs, the mascerated carbon and the high-density graphite throats, can be lightened in weight. The lock-wire nozzle-to-case joint, although reliable, will be changed to a regular threaded attachment because of greater ease of handling and a probable decrease in out-of-tolerance buildup in the nozzle alignment. It is also planned to increase the load of high-temperature pyrotechnic booster material in the squib from 160 to 500 mg in occurrence with the change to Alclo pellets because of the known difficulty

in igniting aluminum potassium-perchlorate pellets as compared to boron-potassium nitrate pellets. The basket-to-basket-ring attachment will be redesigned so that adequate retention of the basket will be provided at least through the ignition phase.

c. Nozzle test results. An all-plastic compression molded nozzle has been designed, fabricated, and tested on the *Syncom* heavywall motor. The nozzle and exit cone were molded in one piece using mascerated carbon cloth with a phenolic resin binder (Fig 5). The nozzle was designed so that approximately one-third of it is submerged in the motor. A backup nozzle was designed, fabricated, and tested in case the original configuration failed. A graphite entrance and throat section was utilized, with all other dimensions and materials remaining the same (Fig 6).

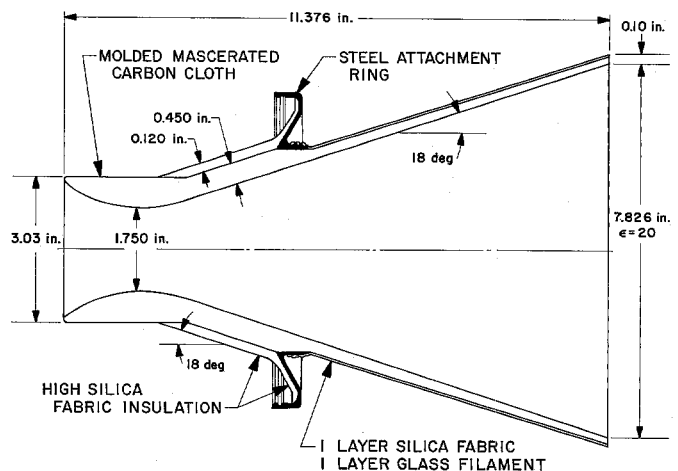


Figure 5. Nozzle configuration with all-plastic throat and exit cone

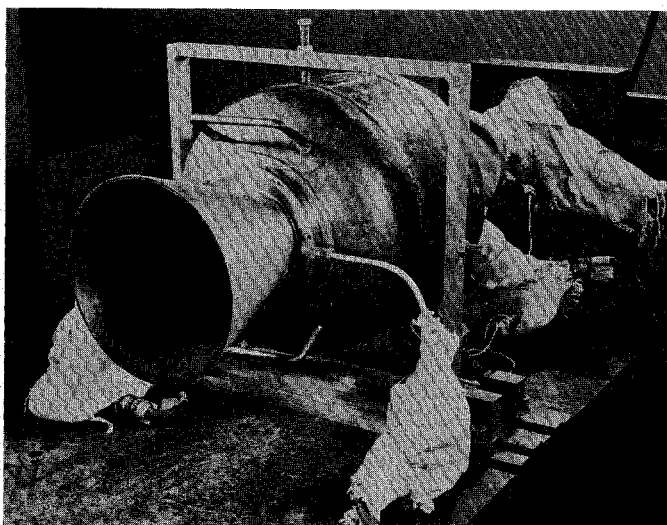


Figure 4. Test setup for heavy wall firing

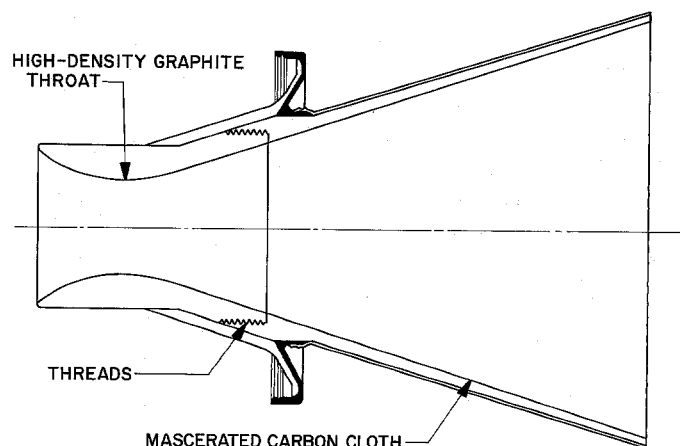


Figure 6. Nozzle configuration with graphite throat

Table 2. Syncom nozzle test results

Nozzle	Run	Configuration	Throat diameter		Burning time, sec	Average pressure, psia	Erosion rate, mils/sec	Area throat change, %
			Initial, in.	Final, in.				
SY-1	X-1152	All plastic	1.750	1.805	17.95	202	3.06	6.2
SY-2	X-1166	All plastic	1.750	1.778	18.00	197	1.55	3.0
SY-3	X-1177	Graphite throat	1.750	1.7515	17.95	198	0.09	0.2
SY-4	X-1185	Graphite throat	1.750	1.752	20.00	150	0.10	0.25

To date, four nozzles have been tested at sea level conditions. These included two all-plastic nozzles, and two nozzles with graphite throats. Table 2 presents the test results. None of the nozzles failed in any respect. The all-plastic nozzles exhibited a low erosion rate. The graphite throats essentially showed no erosion, but the graphite increased the nozzle weight approximately 100 gm. Figure 7 is a cross section view of the first all-plastic nozzle tested. Figure 8 is a view of the nozzle, after firing, showing the outside surface condition. Figure 9 is a sketch which shows the char depth exhibited by the all-plastic nozzle.

As shown by Figures 7 and 8, the steel attachment ring did not receive enough heat to cause any concern. Based on these firings, a redesign of the nozzle has been initiated which will reduce the thickness of the compression molded all-plastic nozzle.

3. Structural Testing

An extensive program is under preparation for the structural testing of the rocket motor. The structural

adequacy of the case and the thrust attachment to the satellite will be tested for the dynamic environment generated by the third stage (ABL 248 solid rocket motor) of the *Thor-Delta* vehicle by static testing in a centrifuge. Tatnall strain gages, accelerometers and stress-coating will be used. Two inert motors will be available for these experiments. Three unloaded cases will be used for hydro-burst tests to determine the margin of safety in the case during motor operation. Dynamic tests of an inert loaded motor will be conducted at the JPL environmental laboratory to simulate the boost vibration environment. Gain factors at specific frequency ranges, as determined by

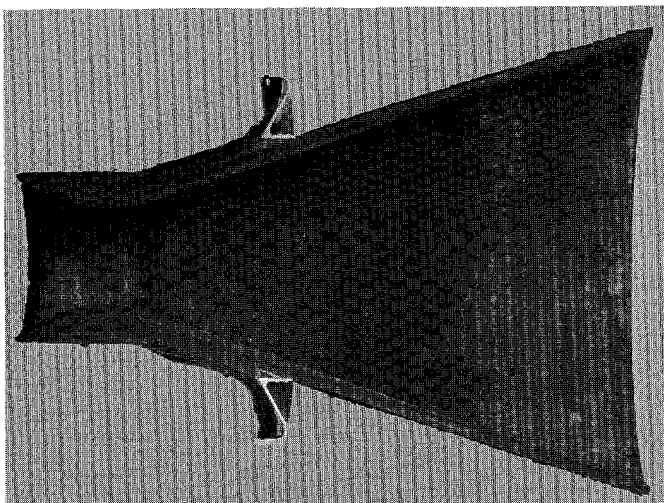


Figure 7. Post-firing sectioned nozzle (Run X-1152)

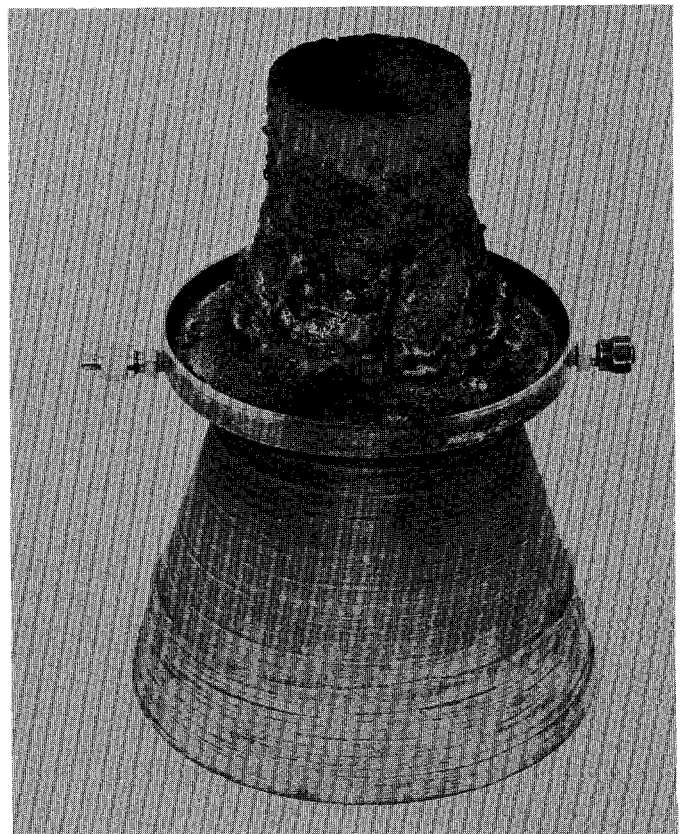


Figure 8. Nozzle after test (Run X-1152)

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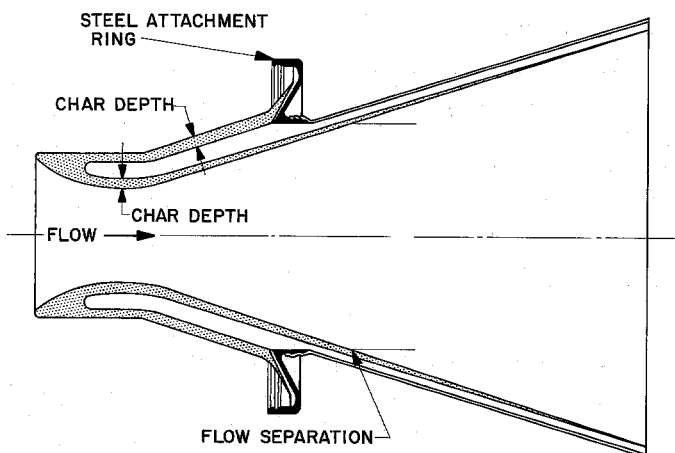


Figure 9. All-plastic nozzle showing char depth

Hughes Aircraft Company from the vibration tests of the *Syncom* satellite with an inert motor, will be applied to the dynamic tests.

4. Flight-Weight Case Development

The *Syncom* motor case was designed to be fabricated from AISI 410 stainless steel sheet material. Sheet metal construction was selected because it offered the best control of wall thickness at minimum cost. Thickness measurements on actual hydroformed heads vary only ± 0.0005 in. By adding this to the original ± 0.001 -in. tolerance of the starting sheet, it has been possible to control the thickness to about $\pm 10\%$ of the nominal 0.014-in. wall.

One disadvantage to sheet metal construction is the problem of attaching the mounting points without introducing a zone of high localized stress. Brazing with a

ductile alloy has been evaluated to overcome this problem. A 35% gold-65% copper alloy was found to possess the best characteristics of those alloys tested. Tension tests on simulated joints demonstrated a shear strength in excess of 50,000 psi. Little difficulty with this type of joint is expected.

A simultaneous brazing and heat-treating cycle has been investigated with excellent results. Cases treated this way have fully hardened, and show very little distortion. ASTM notched tensile tests in 0.025-in. samples given a simulated braze and heat treat cycle have consistently had edge-notched-to-yield-strength-toughness ratios of 1.16 or greater at a yield strength of 156,000 psi, indicating that AISI 410 steel at the design strength level has sufficient toughness to perform in a predictable manner.

5. Design Integration

Moments of inertia and weight and center-of-gravity locations of the rocket motor for the before and after burnout conditions have been calculated for preliminary examination of the dynamic compatibility of the motor with the rest of the satellite payload. Quotations are now being sought for weight and balance units which will enable the empty rocket motor to be balanced within 5 oz-in.² about the nozzle thrust centerline.

Hughes has indicated a potential decrease of space available between the forward dome of the rocket motor and the aft end of the center cylindrical electronics package. The rocket motor ignition power cabling and connectors are to be located in this space. A design study is now underway concurrent with the redesign of the igniter in an attempt to minimize the required spacing.

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